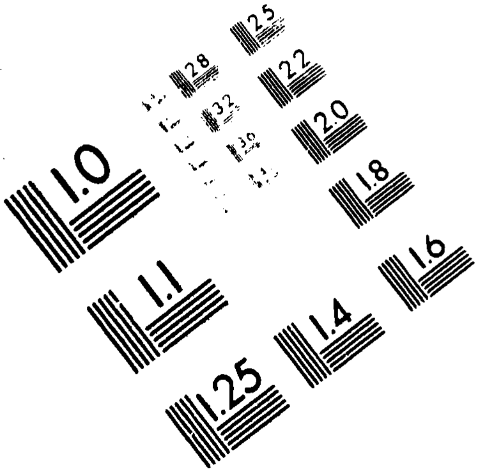

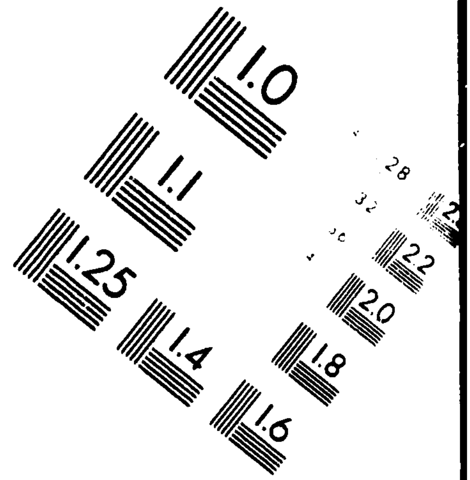

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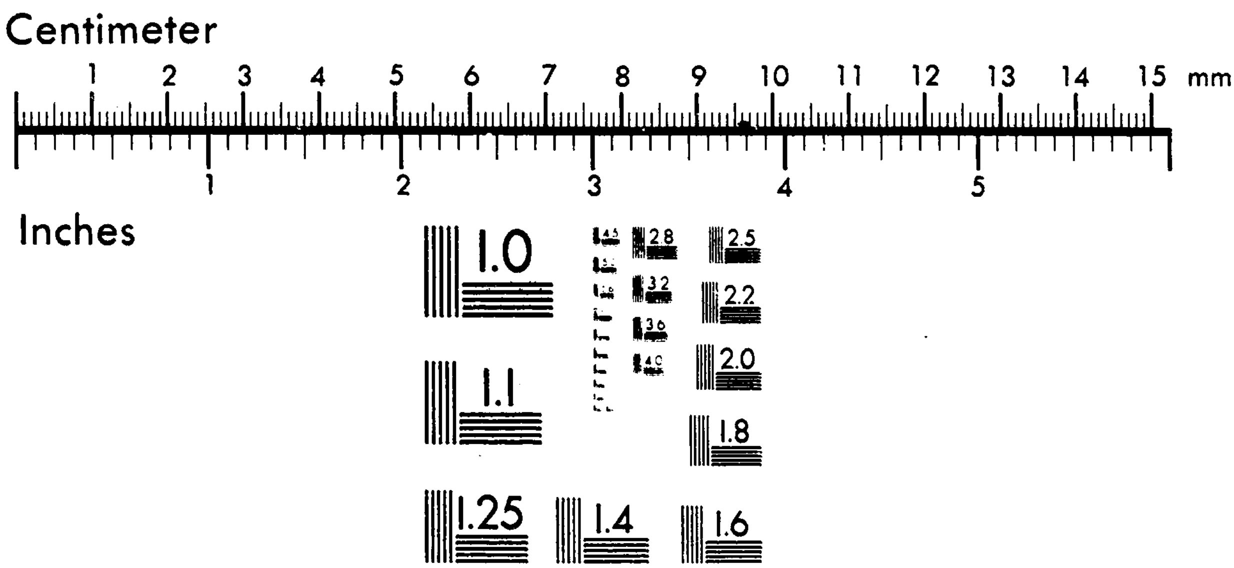

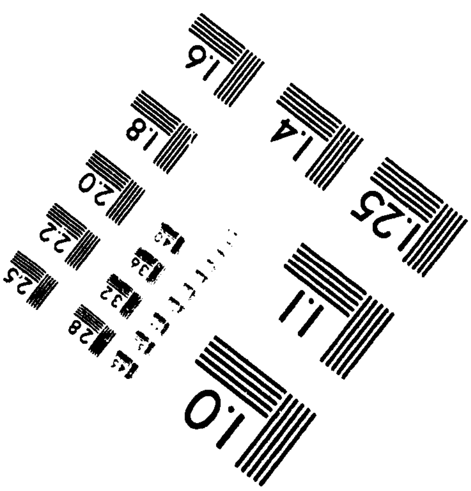

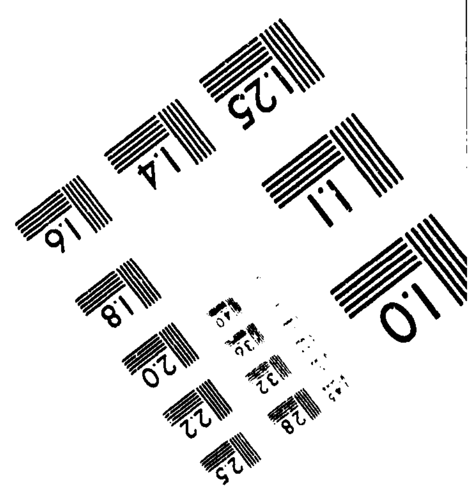

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SP-100 Program

Technical Information Report

SP-100 Planetary Mission/System Preliminary Design Study

Final Report

Edited by

Ross M. Jones

February 1986

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

SP-100 PROGRAM

DISCLAIMER

Department of Defense
Strategic Defense Initiative Organization
Washington, D.C.

Department of Energy
Office of Nuclear Energy
Washington, D.C.

National Aeronautics and Space Administration
Office of Aeronautics and Space Technology
Washington, D.C.

Jet Propulsion Laboratory
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ABSTRACT

This report contains a discussion on many aspects of a nuclear electric propulsion planetary science mission and spacecraft using the proposed SP-100 nuclear power subsystem. A review of the science rationale for such missions is included. A summary of eleven nuclear electric propulsion planetary missions is presented. A conceptual science payload, mission design, and spacecraft design is included for the Saturn Ring Rendezvous mission. Spacecraft and mission costs have been estimated for two potential sequences of nuclear electric propulsion planetary missions. The integration issues and requirements on the proposed SP-100 power subsystem are identified.

ACKNOWLEDGMENT

Appreciation is expressed to all study participants (listed alphabetically).

STUDY PARTICIPANTS

The following people made major contributions to this study. Unless otherwise noted, all are JPL employees.

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*SAIC = Science Applications International Corporation.

#LeRC = Lewis Research Center.

EXECUTIVE SUMMARY

The purposes of this study are to identify (1) a broad set of nuclear electric propulsion (NEP) planetary mission applications and (2) the requirements that those mission applications place upon a SP-100 power subsystem. Eleven NEP missions are identified (see Table 1). The Saturn Ring Rendezvous mission (see Figures 1 and 2) is selected as the representative mission for the purposes of this study. A spacecraft conceptual design (see Figure 3) was completed. The environment in which the spacecraft would have to operate is defined.

The cost and sequence of two representative sets of missions are determined. The first mission set contains four missions beginning with a 1997 launch of a sample return mission to Mars and ending with the launch of a Saturn ring rendezvous mission in 2006. The total cost of this first mission set is estimated as $4.52B (FY 84 dollars). The second mission set contains five missions beginning with a 1996 launch of a dual sample return mission to Vesta and Wild 2 and ending with the launch of a Pluto orbiter in 2006. The total cost of this second mission set is estimated as $4.0B (FY 84 dollars).

The following items are the major conclusions of this study.

(1) A NEP spacecraft can provide the capability to travel to any body in the solar system and complete the intensive investigation of those bodies, thereby enable a new and exciting era of planetary exploration.

(2) There is substantial science rationale for and interest in several of the many planetary missions that are enabled or significantly enhanced by nuclear electric propulsion.

(3) A feasible overall spacecraft configuration using an SP-100 power subsystem, an electric propulsion subsystem, and a typical planetary instrument payload can be defined using current or projected technical capabilities.

(4) Seven years of full power operation, 100 kW_e, and 3,000 kg are acceptable goals for a SP-100 power subsystem for NEP planetary missions.

(5) The radiation environment is the single most challenging feature of the SP-100 for planetary spacecraft design and integration.

(6) The SP-100 power subsystem (as defined by its present baseline specifications) would be compatible with NEP planetary missions if options being considered for the baseline power subsystem were included. The adjustments to the baseline SP-100 power subsystem specifications that would make it compatible with planetary missions are (1) a longer life of about 12 years, (2) up to five years of dormancy, (3) the ability to throttle the reactor down in power about a factor of ten and back to full power at least for five full cycles, (4) lower reactor produced radiation, and (5) the ability to survive a more severe meteoroid environment.

Table 1. Potential NEP(1) Missions

Mission	Mission Time (years)	Payload (kg)
Neptune Orbiter	11.2	1500
Saturn Ring Rendezvous	8.8	1500
Uranus Orbiter	8.8	1500
Jupiter Tour	5.1	1500
Pluto/Charon Orbiter	10.9	1500
Mars Sample Return	5.2	1000(2)
Asteroid Vesta Sample Return	5.2	1500(3)
Comet Tempel 2 Sample Return	7.3	1000(4)
Comet and Asteroid Rendezvous	6.4	2300(5)
Venus Orbiter	1.7	2500
Mercury Orbiter	2.6	2000

(1) 100 kW_e 5300 sec.

(2) 5800 kg jettisoned at Mars.

(3) 2000 kg left at Vesta.

(4) 500 kg left at Tempel 2.

(5) 1000 kg left at Lumen.

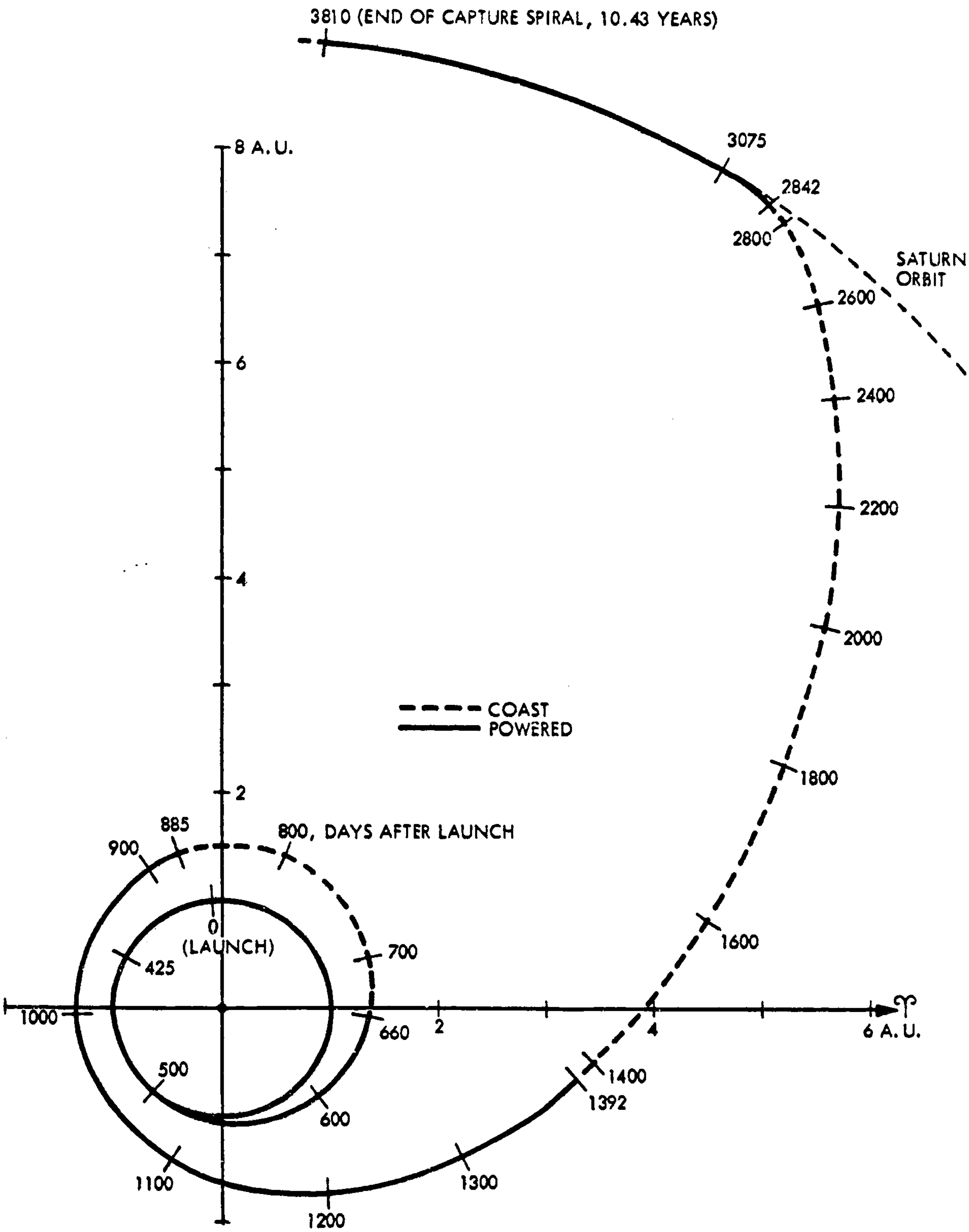

Figure 1. Reference Heliocentric Saturn Ring Rendezvous Mission Trajectory

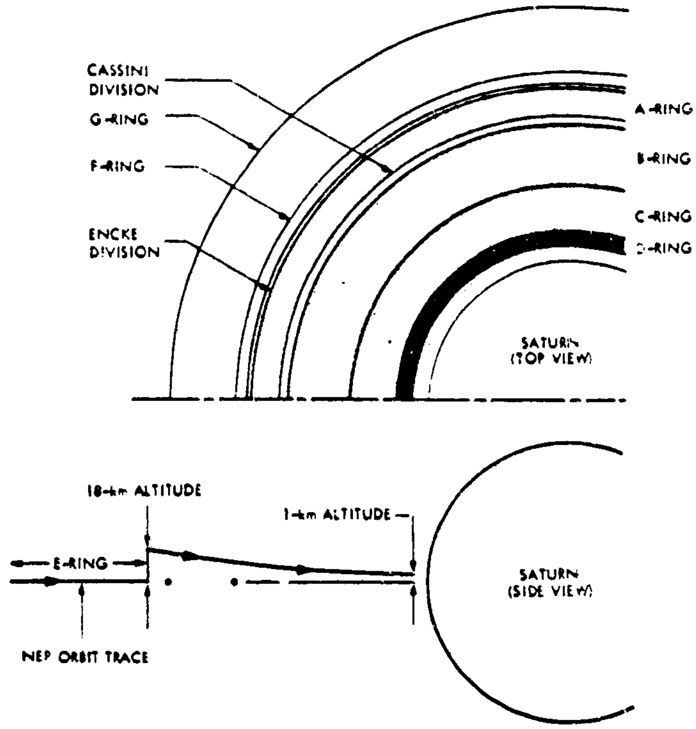

Figure 2. Nominal Geometry (Not to Scale) of the Saturn Ring Rendezvous Survey Mission Phase

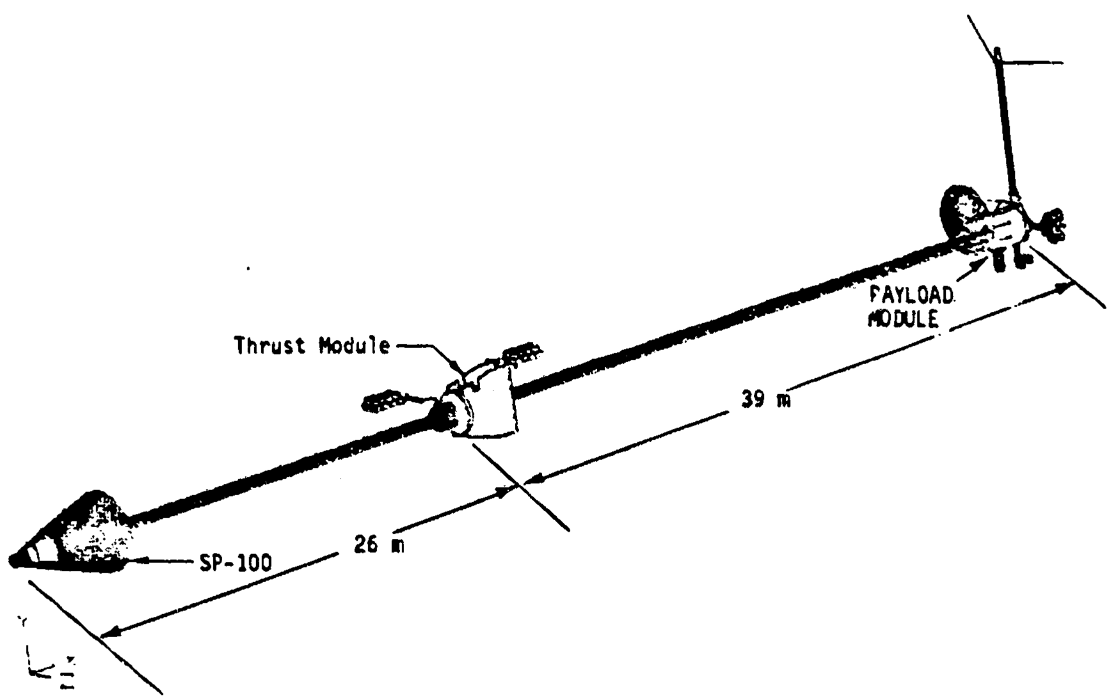

Figure 3. Nuclear Electric Propulsion Saturn Ring Rendezvous Spacecraft Configuration

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SECTION 1

INTRODUCTION AND KEY ASSUMPTIONS

1.1 INTRODUCTION

This study was performed by the Mission Analysis and Requirements element of the SP-100 Project. The SP-100 Project is developing the technology for a 100-kWe class (50-1000 kWe) nuclear electric power supply. The roles of Mission Analysis and Requirements within the SP-100 Project are to identify mission applications for the SP-100 power subsystem and to identify and define the requirements that these potential mission applications place upon the SP-100 power subsystem. Potential mission applications for the SP-100 power subsystem include military, space station, commercial/public service, and space science (including planetary science). Although planetary missions could use the power from a SP-100 for nonpropulsive uses such as high power science and data transmission, the most attractive use of SP-100 power for planetary mission applications is for electric propulsion. Nuclear electric propulsion (NEP) has long been recognized as a means by which a whole class of planetary missions can be performed that either cannot be performed without NEP or can only be performed at a reduced level and scope.

The purposes of this study are to identify (1) a broad set of NEP planetary mission applications and (2) the requirements that these mission applications place upon a SP-100 power subsystem. NEP planetary missions are identified by first reviewing the general planetary science objectives as outlined by the National Research Council's Space Science Board's Committee on Planetary and Lunar Exploration and by NASA's Solar System Exploration Committee. The NEP missions that could contribute to meeting the objectives, as defined by the above committees, are defined. A single NEP mission, which represented the whole set of NEP missions, and whose mission characteristics placed upon the SP-100 power subsystem requirements that would be as stressing or more stressing than any other potential NEP mission, has been selected. The Saturn Ring Rendezvous Plus Radar (SRRPR) mission is selected as the single, stressing mission. As this report shows, a NEP SRRPR mission may or may not be feasible; but by attempting to design and build a SP-100 power subsystem to specifications coming from this stressing mission, the study participants are confident that the SP-100 power subsystem will be able to successfully fulfill the requirements of many easier or less stressing NEP planetary mission applications.

In order to define requirements on the SP-100 power subsystem, a SRRPR mission and system conceptual design has been completed. The mission and system design focused on those areas where the major requirements on the SP-100 are likely to originate. General mission timelines, including power requirements, are provided in this report. The spacecraft system conceptual design includes (1) the electric propulsion subsystem and its electrical interface with the SP-100, (2) the attitude and articulation control subsystem, (3) the telecommunications subsystem, (4) a study of science instrument radiation tolerances, and (5) a definition of a feasible configuration with mass properties. The definition of a nominal 100-kWe

SP-100 power subsystem is given, provided to the study by the System Definition element of the SP-100 Project. The meteoroid, radiation, and Saturn ring particle environments through which the SP-100 power subsystem must pass are also defined. The SRRPR mission cost was estimated, and a NEP mission plan formulated.

This study was performed between April and December 1984, although some work was accomplished prior to April 1984. The work reported here is based upon the information available during the period previously mentioned. NEP planetary missions are but one of four general classes of potential SP-100 missions. The requirements identified in this study may be in varying degrees of conflict with those from other mission classes. These conflicts will need to be resolved by the SP-100 Project based upon its perception of mission applications priority and technical feasibility.

1.2 KEY ASSUMPTIONS

The important key assumptions utilized by the study team participants as the basis of this report are the following:

(1) The science rationale for planetary missions that are enabled or significantly enhanced by NEP is taken from documents produced by the Space Science Board of the National Academy of Sciences and by their subcommittee, specifically the Committee on Planetary and Lunar Exploration (COMPLEX).

(2) The NEP system concept that is presented in this report uses a 100-kWe, 2500- to 3000-kg power subsystem and a 1500- to 2500-kg, 5300-sec, 30-cm mercury ion propulsion subsystem.

(3) The NEP system is compared to solar electric propulsion (SEP) and chemical propulsion for several planetary missions on the basis of trip time and initial mass both for a constant payload. The SEP system uses the "standard" technology of about 30 kWe and 3000 sec beginning of life. The retrosystem for both SEP and chemical propulsion has a specific impulse of either 300 or 370 sec. Aerocapture is also included as an option.

(4) The Centaur G-prime is the upper stage used to inject the SEP and chemical propulsion missions.

(5) The nominal 29,500-kg payload shuttle is the launch vehicle for all options.

(6) The reactor in the NEP system is not started until the system is delivered to 700 km. The NEP system is delivered to a 700-km circular orbit by a small chemical propulsion unit.

(7) The Galileo instruments are selected as representative of planetary science instruments and are used for a study of instrument sensitivity to SP-100 radiation levels.

(8) The Saturn Ring Rendezvous (SRR) mission is selected as representative of the most difficult NEP planetary mission and is used for detailed mission and system design.

(9) The maximum required data rate from Saturn for the Saturn Ring Rendezvous mission is 268.8 kbps.

(10) The calculated meteoroid environment for the Saturn Ring Rendezvous mission includes the Earth escape, heliocentric phase, and the capture at Saturn.

(11) The calculated radiation environment for the Saturn Ring Rendezvous mission includes Van Allen belts at Earth, free space solar flares, and the Saturn environment.

(12) The radiation shielding calculations for reactor-produced radiation includes the use of the mercury propellant as a shield.

(13) The NEP Saturn Ring Rendezvous trajectory takes the spacecraft through the E-, F-, and G-rings and above the A-, B-, C-, and D-rings.

(14) A 30% contingency is added to all mission cost estimates.

(15) Only the recurring costs of the ion propulsion subsystem are included. It is assumed that the development costs are borne by a previous SEP mission.

(16) Only the recurring costs of the SP-100 power subsystem are included. It is assumed that the development costs are borne by a previous mission.

(17) The nominal recurring cost of the SP-100 power subsystem is $45M (FY 84 dollars).

(18) Shuttle launch costs are not included in the mission costs as is the custom for planetary missions.

SECTION 2

PLANETARY EXPLORATION SCIENCE OBJECTIVES

The purpose of this section is to present a summary of the rationale for planetary science space missions. The fundamental rationale for these missions is the fulfillment of science objectives. As will be seen in the next section, some of these science objectives may be met by missions that are enabled or enhanced by nuclear electric propulsion (NEP).

The United States of America committed itself to the fulfillment of science objectives in the space environment by the enactment of the National Aeronautics and Space Act of 1958. The National Aeronautics and Space Act of 1958, in its Declaration of Policy and Purpose, lists two of eight major objectives as follows:

- The expansion of human knowledge of phenomena in the atmosphere and space.
- The establishment of long-range studies of the potential benefits to be gained from the opportunities for, and the problems involved in, the utilization of aeronautical and space activities for peaceful and scientific purposes.

The responsibility for the development of science objectives and goals and the strategy through which they may be fulfilled has been undertaken by the Space Science Board of the National Academy of Sciences. The Space Science Board has been active in the development of science objectives and strategies for many years (References 2-1 to 2-10). In the early years (References 2-1 to 2-6), recommendations for specific missions were developed. In later years (References 2-7 to 2-10), only general sciences objectives and strategies have been defined. Recently (Reference 2-11), a NASA advisory committee has taken the planetary science objectives and strategies developed by the Space Science Board and has again recommended specific planetary missions.

The Space Science Board recognizes three major divisions of space science: (1) Space Astronomy and Astrophysics, (2) Solar and Space Physics, and (3) Planetary and Lunar Exploration. Space Astronomy and Astrophysics, for the most part, do not define science objectives that call for missions which could use NEP. Solar and Space Physics have defined one basic principle for their science strategy (Reference 2-10), which may lead to missions enabled or enhanced by NEP.

> "The objectives of solar-system space research are to understand the physics of the sun; the heliosphere; and the magnetospheres, ionospheres, and upper atmosphere of the Earth, other planets, and comets."

The area of Planetary and Lunar Exploration has the most science objectives that can be met by NEP missions. The overall science statement for Planetary and Lunar Exploration was presented in Reference 2-6 as follows.

> "The primary scientific goals in investigating the solar system are to determine the composition, structure and environment of the planets and their satellites in order to define the present morphology and dynamics of the solar system and with the purpose of making major steps in understanding the processes by which the planets formed from the solar nebula and how they have evolved with time and how the appearance of life in the solar system is related to the chemical history of the system. The investigation of the interplanetary and interstellar medium is considered an intrinsic part of such an endeavor."

Reference 2-7 presented the following science objectives for the exploration of Mars, Venus, Mercury, and the Moon.

> "In summary, the primary objectives in order of scientific priority for the continued exploration of Mars are (1) the intensive study of local areas (a) to establish the chemical, mineralogical, and petrological character of different components of the surface material, representative of the known diversity of the planet; (b) to establish the nature and chronology of the major surface forming processes; (c) to determine the distribution, abundance, and sources and sinks of volatile materials, including an assessment of the biological potential of the Martian environment, now and during past epochs; (d) to establish the interaction of the surface material with the atmosphere and its radiation environment; (2) to explore the structure and general circulation of the Martian atmosphere; (3) to explore the structure and dynamics of Mars's interior; (4) to establish the nature of the Martian magnetic field and the character of the upper atmosphere and its interaction with the solar wind; (5) to establish the global chemical and physical characteristics of the Martian surface."

> "The primary objectives of the exploration of Venus . . ., in order of importance, are (1) to obtain a global map of the topography and morphology of its surface at sufficient resolution to allow identification of the gross processes that have shaped the surface, (2) to determine the major chemical and mineralogical composition of the surface material, (3) to determine the concentrations of photochemically active gases in the 65-135 km altitude region, and (4) to investigate the physical and chemical interactions of the surface with the atmosphere and the composition and formation of atmospheric aerosols."

> "The primary planetary objectives in the exploration of Mercury . . . are to determine the chemical composition of the planet's surface on both a global and regional scale, to determine the structure and state of the planet's interior, and to extend the coverage and improve the resolution of orbital imaging."

"The primary scientific objectives for exploration of the Moon by spacecraft . . ., in order of importance, are (1) to determine the chemistry of the lunar surface on both a global and regional scale; (2) to determine the surface heat flow on both a global and regional scale; and (3) to determine the nature of any central metallic core in the Moon."

Reference 2-9 presented the following science objectives for the exploration of Comets and Asteroids.

"The primary scientific objectives of comet exploration . . . are, in order of priority: (1) To determine the composition and physical state of the nucleus (determination of the composition of both dust and gas is an important element of this objective); (2) To determine the processes that govern the composition and distribution of neutral and ionized species in the cometary atmosphere; and (3) To investigate the interaction between the solar wind and the cometary atmosphere. In view of the apparent diversity of comets, it is important that comparative measurements be made, including measurements of objects in different stages of evolution. Furthermore, it is important to observe the changing state of the nucleus and coma of a comet during perihelion passage."

"The primary scientific objectives for the exploration of asteroids are, in order of priority:

1. To determine their composition and bulk density;

2. To investigate the surface morphology, including evidence for endogenic and exogenic processes and evidence concerning interiors of precursor bodies; and

3. To determine the internal properties, including states of magnetization of several carefully chosen asteroids selected on the basis of their diversity."

The Space Science Board has not developed an updated set of science objectives for the exploration of the outer planets, i.e., those beyond Mars. In lieu of the Space Science Board recommendations, Reference 2-12 presented the following science objectives for outer planet exploration.

"Explore:

(1) The primary body: its internal structure, surface and atmosphere.

(2) The satellites of each primary: their internal structures, surfaces, and any atmospheres they might possess.

(3) Ring structures that might exist about the primary (one could possibly consider this a subclass of satellites).

(4) The magnetic field of the primary, the magnetosphere of the system, and the magnetospheric interactions between trapped radiation and the primary and its satellites.

(5) The interplanetary environment that exists beyond the orbit of Saturn."

The Space Science Board defined the types of planetary exploration missions as well as the science objectives. Reference 2-6 presents these mission types as follows.

> "The investigation of any solar-system object can be divided into three categories: reconnaissance, exploration, and intensive study. As the first step of our qualitative scale of progress on a given planet, we may speak of reconnaissance in which major characteristics are first sought and identified. Reconnaissance tells us qualitatively what the planet is like and provides enough information about the character of the planet and its environment to allow us to proceed to the stage of exploration of the planet. Exploration seeks the systematic discovery and understanding of the processes, history, and evolution of the planet on a global scale. In the final step, that of intensive study, sharply formulated specific problems of high importance are pursued in depth. The sequence of investigations should follow the order of reconnaissance, exploration, and intensive study."

The mission types of exploration and intensive study are the most likely to require the propulsion performance that NEP can supply.

Finally, Reference 2-11, has taken the preceding science objectives and recommended a series of planetary exploration missions that are limited in scope, science return, and technical challenge by the present space science funding realities. This "Core Program" will not include NEP missions. However, Reference 2-11 also recognized the need for a program beyond the "Core Program" as follows:

> "Therefore, the Committee recommends that the Core program be augmented at the earliest opportunity by missions of the highest scientific priority that are also significantly more technically challenging than those of the Core program."

It is to be expected that many of the missions in the potential "Augmented Program" may be enabled or enhanced by NEP.

SECTION 3

NUCLEAR ELECTRIC PROPULSION AND PLANETARY EXPLORATION

3.1 WHY NUCLEAR ELECTRIC PROPULSION?

The preceding section discussed the where, why, and what of planetary exploration; i.e., where we want to go, why we want to go there, and what we want to do/learn. This section addresses the how, i.e., how can we get to where we want to go. Specifically, this section presents a rationale for why nuclear electric propulsion (NEP) is an excellent mode of interplanetary transportation.

Besides NEP, solar electric propulsion (SEP) and chemical propulsion are transportation options for planetary missions. For orbiter missions, aerobraking/aerocapture may be used to reduce the propulsion system requirements for the orbit insertion maneuver. Gravity assists from Earth and Jupiter may also be used as a means of interplanetary transportation. The Voyager missions are taking advantage of the planetary alignment of Jupiter, Saturn, Uranus, and Neptune, which occurs approximately once every 180 years, in order to use a gravity assist from the preceding planet in order to travel to the next. Without aerocapture/aerobraking and a gravity assist from Jupiter, flight times to the outer planets are unacceptably long.

As an example, a far outer planet mission using a Jupiter gravity assist (available every 12 years) would require a flight time of 12.5 to 14 years (depending on final mass margin and launch date) to deliver the Galileo orbiter with a probe to Uranus and 20 to 21.3 years for Neptune. These flight times are for a single shuttle launch with a Centaur G-prime upper stage and a final orbit of 3 R_p x 100 days. SEP can reduce these trip times, but the power from solar arrays is reduced by a factor of 100 at Saturn relative to what it is at Earth. Therefore, SEP is useful only for the initial part of a trip to the outer planets and not for orbit insertion.

Figure 3-1 presents a comparison of trip times for outer planet orbiters under the constraints of a fixed payload and a single shuttle launch. It is clear that NEP is superior. Even SEP is better than the all chemical propulsion options. In Figure 3-1, the top of the chemical propulsion bars indicate the trip time using both Earth and Jupiter gravity assists. For Neptune and Uranus, the chemical propulsion performance has been augmented by aerocapture at the planet, but even so NEP is superior. Figure 3-1 limited the mission initial mass to less than one shuttle payload. Figure 3-2 removes this constraint but retains the less than 10-year trip time for an example (Neptune orbiter) mission. Figure 3-2 presents a comparison of NEP to alternative (some very advanced) propulsion options. The basis of comparison is the initial mass in low Earth orbit (LEO). NEP is again superior.

Figure 3-3 compares NEP to some near term alternatives, such as on-orbit assembly of Centaur G and G-primes and a hypothetical orbit transfer vehicle (OTV). Again the mission candidate is the Neptune orbiter (11.1-year flight time, 1500-kg payload) under the assumptions of a fixed flight time and payload. Figure 3-3 also includes aerocapture insertion, higher performing

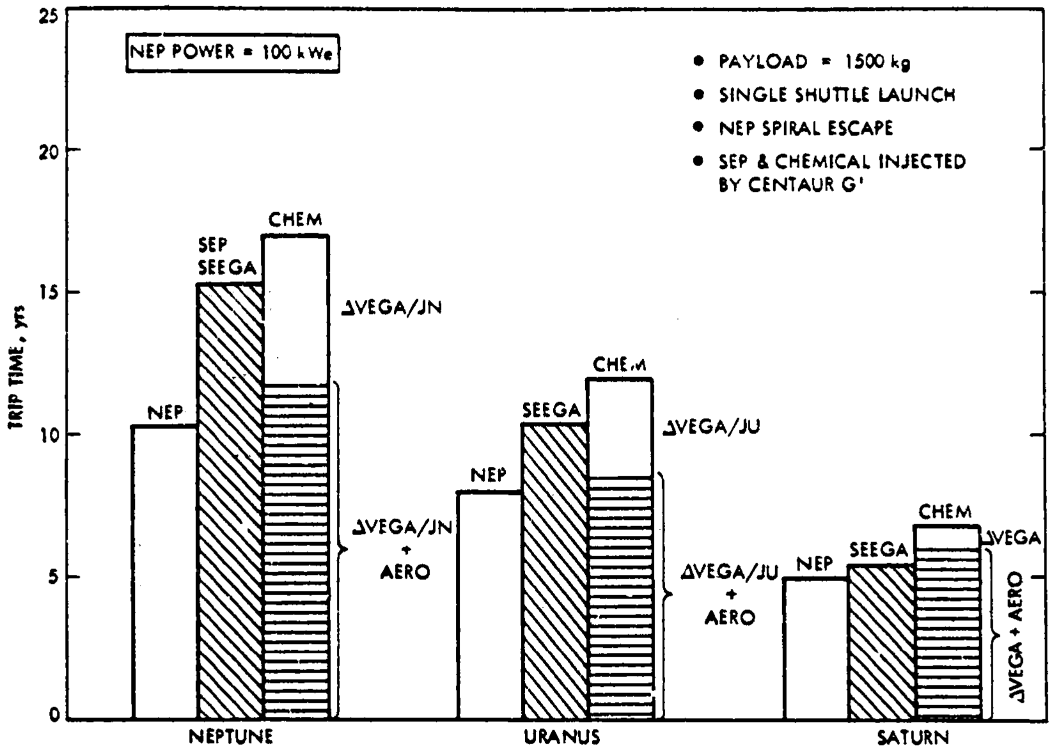

Figure 3-1. Outer Planet Orbiter Mission Trip Times

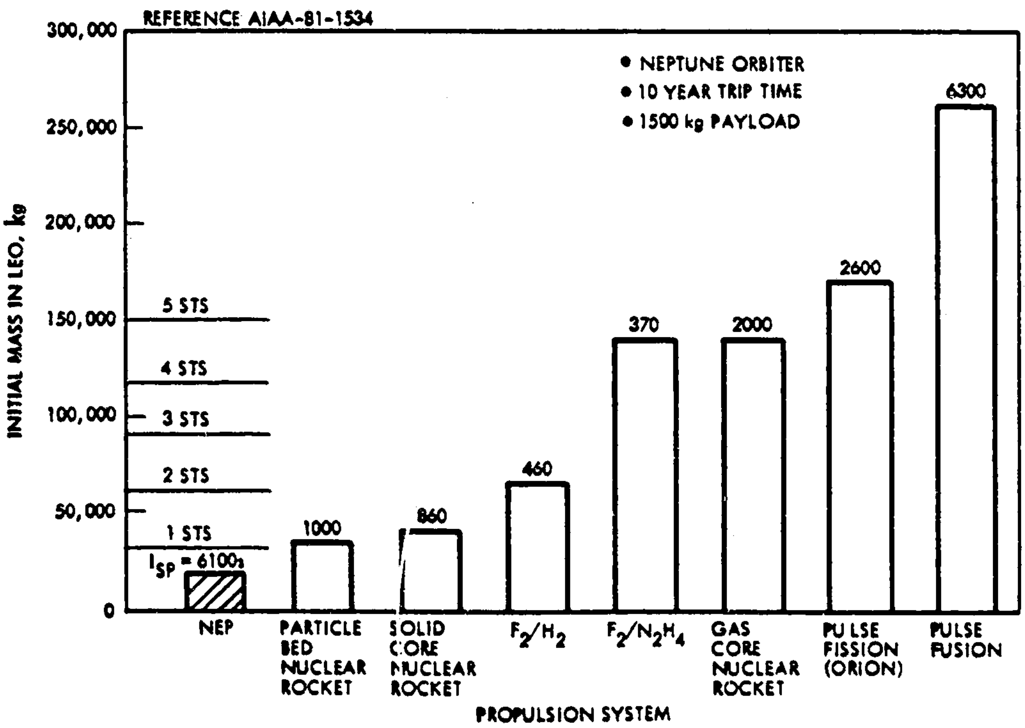

Figure 3-2. Spacecraft Initial Mass Versus NEP and Advanced Propulsion Systems for the Neptune Orbiter Mission

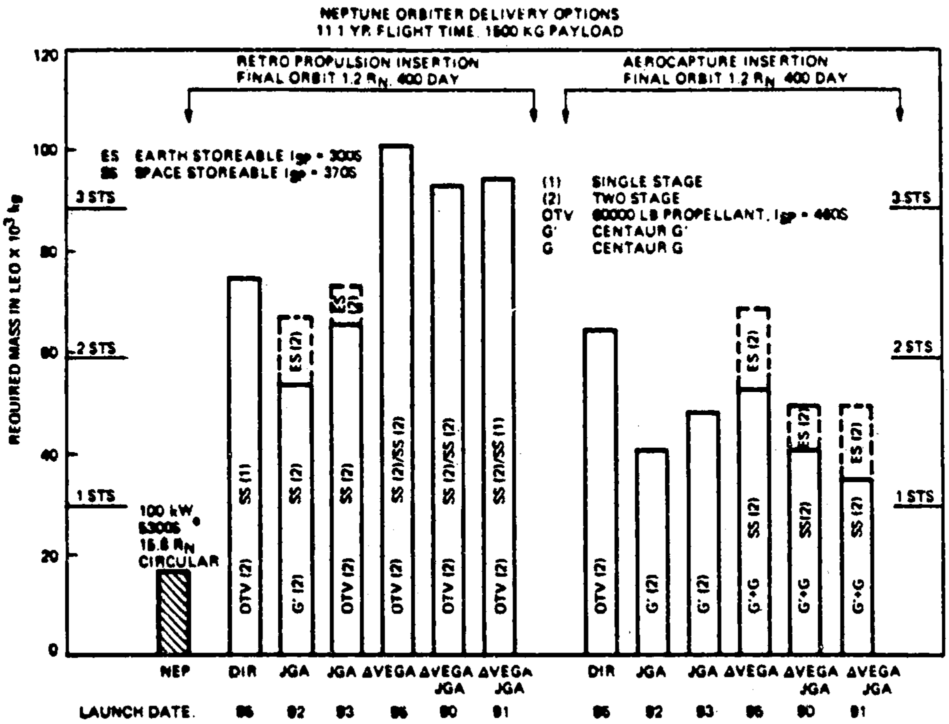

Figure 3-3. Spacecraft Initial Mass Versus NEP and Conventional Propulsion Systems Plus Aerocapture for the Neptune Orbiter Mission

space storable propellants, and Jupiter and Earth gravity assists. Based upon the figure of merit of initial mass in LEO (shuttle launches), NEP is superior.

A recent study (Reference 3-1) determined the performance of NEP, SEP, and chemical propulsion for sample return missions to three comets: Encke, Tempel 2, and Wild 2. Some of the results of this study are presented in Table 3-1. The flight modes are: (1) chemical propulsion using two shuttle launches and the on-orbit assembly of two Centaur upper stages and either an Earth gravity assist trajectory (ΔVEGA) or a direct trajectory, (2) SEP using a single shuttle launch and a Centaur upper stage, and (3) NEP using a single shuttle launch. The basis for comparison was both flight time and mass margin for a given payload. The mass margin is the difference between the propulsion system capability and the mission injected mass requirement. A negative mass margin indicates that the mission is not possible. Table 3-1 again presents the superiority of NEP to chemical propulsion. For these comet sample return missions both SEP and NEP have advantages. Although not clearly superior compared to SEP for these missions, Table 3-1 does demonstrate the applicability of NEP for comet sample return missions.

The preceding paragraphs have presented the rationale for NEP as the favored transportation mode for outer planet orbiters and as an attractive option for comet sample return missions in the context of a trip time,

Table 3-1. Summary of Comet Sample Return Flight Mode Options

		Encke		Tempel 2		Wild 2	
Flight Mode	Upper Stage	Mass Margin (kg)	Flight Time (yr)	Mass Margin (kg)	Flight Time (yr)	Mass Margin (kg)	Flight Time (yr)
Chemical--ΔVEGA	G'+ G	No Mission		1590	11.1	Not Considered	
Chemical--Direct	G'+ G'	No Mission		No Mission		440	8.9
SEP	G'	950	6.1	980	5.0	1530	6.0
NEP	None	5240	7.0	7590	6.7	6130	7.0

payload, and initial mass in LEO. Besides the trip time and payload rationale, NEP can provide other mission benefits such as (1) a large amount of power at the target, which can support high power science and high data rate transmission, and (2) large mass margins, which can accommodate system mass growth without expensive redesigns.

3.2 POTENTIAL MISSIONS

3.2.1 System Parameters Assumptions

This subsection presents a brief summary of the planetary NEP missions that have been examined together with estimates of performance. The planetary missions to be examined were studied over a period of several years and assumed system parameters, namely system mass, that is not completely compatible with the system concepts presented in the rest of this report. The particular set of power and propulsion system parameters used in generating the mission designs presented in this subsection are detailed in Table 3-2, and assume 2500 kg for the power subsystem mass and a total of 125 kg per thruster for the propulsion subsystem. Two propulsion or thruster system concepts were examined; the first, corresponding to the system concept used in other parts of this report, utilized a thrust system containing 16 thrusters and weighing 2000 kg. The second thruster concept utilized a 12-thruster array with a mass of 1500 kg. This latter subsystem was used for those missions where the full power operating time was significantly less than that required for the 16-thruster subsystem. In all of the cases in this subsection, the same thruster specific impulse and efficiency were used, and the maximum propellant loading for the above subsystem concepts was 10,090 kg for the 16-thruster subsystem and 7570 kg for the 12-thruster subsystem.

Table 3-2. Power and Propulsion Subsystems Parameters

Power Supply (Reactor) Output:	100.0 kWe	
Line Losses:	4.0 kWe	
Housekeeping Power:	0.5 kWe	
Mass of Power Subsystem:	2500 kg	
Net Propulsion Subsystem Input Power:	95.5 kWe	
Propulsion System Technology Level:	30-cm Mercury Ion Thrusters	
Number of Operating Thrusters:	5	
Input Power per Thruster:	19.1 kWe	
Effective Specific Impulse:	5300 seconds	
Combined Thruster/PPU Efficiency:	77.3 percent	
Thrust Level per Thruster:	0.57 newtons	
Total Thrust Level:	2.84 newtons	
Assumed Thruster Lifetime:	20,000 hours at full power	
Number of Redundant Thrusters:	25 percent	
Total Number of Thrusters:	16.0	12.0
Number Required for Lifetime:	12.8	9.6
Effective Redundant Thrusters:	3.2	2.4
Maximum Propellant Loading:	10,090 kg	7570 kg
Maximum Propulsion Time:	5.84 yr	4.38 yr
Thruster Subsystem Mass:	2000 kg	1500 kg
Total Propulsion Subsystem Mass:	4500 kg	4000 kg

3.2.2 Mission Scenarios

Two Earth escape concepts were examined for most of the missions presented in this subsection. The first concept involved a low-thrust spiral transfer phase to achieve escape from Earth. The second concept achieved Earth escape using a separate high energy injection stage that would be jettisoned after escape. There are both advantages and disadvantages associated with each concept. The first concept, using a low-thrust spiral escape, has the advantage in that it requires only a single Space Transportation System (STS) launch; but has a disadvantage in that it requires, in many cases, more than a year to achieve escape from the Earth. This long escape time translates into increased full power reactor and thruster operating time and requires the addition of about four of the 16 thrusters just for the Earth escape phase. The second concept achieves a high thrust escape to a slightly positive injection energy and, thus, enables a mission with a propulsion subsystem that requires fewer than the 16 thrusters needed for the first concept. The disadvantage of this second concept is that it would likely need to be supported by more than one STS launch in order to orbit both the high thrust chemical stage and the NEP vehicle and payload. This second concept does have the benefit in that the reactor would not be powered up until after the vehicle had achieved escape energy from Earth. The mission analysis has assumed that an escape utilizing the spiral escape mode commences from a circular orbit altitude of 700 km while the high thrust chemical escape is assumed to start from a 400-km altitude circular orbit.

In both concepts, the escape phase is followed by an optimized heliocentric transfer phase, which may or may not contain coast phases and a capture phase at the target. Several mission options were also examined for the target capture phase, which involved either high-thrust capture or low-thrust spiral capture. These options are further described in the individual mission descriptions to be presented. Capture for asteroid and comet missions can be accomplished with the electric propulsion system, since it involves a rendezvous with a low mass body. The same conditions described above will also apply for the several sample return missions which are included. In these missions, capture at Earth return may be achieved either using a low-thrust capture spiral trajectory in which the entire NEP vehicle and return payload would be returned to the nuclear safe orbit* (NSO) or using a separable chemical retro stage to return the sample payload to an orbit where it could be easily retrieved.

3.2.3 Mission Performance Summaries

A variety of potential planetary missions have been examined using the power and propulsion subsystem concepts described above. These examples cover a range of mission categories most likely to be considered for NEP missions in the next decade or two utilizing SP-100 level technology. These missions consider typical examples for several of the different missions and are intended to indicate performance potential for a particular class of mission using the assumed technology level. Thus, only one sample return mission to an asteroid or a comet is included, although many additional targets are equally accessible with these NEP subsystems. Although many of the missions to be presented are analyzed for a particular launch schedule, the actual launch and arrival dates are not listed to avoid the implication that a particular mission development schedule is being presented. In general, the performance for outer planet missions is relatively insensitive to launch opportunity for a NEP mission, and these missions were studied assuming circular coplanar orbits for both the Earth and the target body. The remainder of the missions investigated were studied using a conic ephemeris for the departure and arrival bodies. Rendezvous trajectories to either a single asteroid or comet were not included in this summary since such missions could be accomplished with either a ballistic or SEP spacecraft system. The NEP subsystem reaches its full potential for those high energy missions that may be impractical using either a chemical system or a SEP system. Therefore, most of the missions being considered are either high energy outer planet missions or high energy sample return missions.

The performance and trajectory characteristics for the planetary missions examined in this study are summarized in the following set of tables. The first, Table 3-3, presents a summary of several of the mission characteristics including total mission time and full power lifetime required from the thruster and power subsystems. Included in this table is an entry denoting whether the mission is accomplished with a thruster array consisting

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*Nuclear safe orbit is defined as an orbit where the spacecraft will not enter the Earth's atmosphere within 300 years. (For the type of spacecraft in this report, the orbit altitude for a nuclear safe orbit would be 700 km.)

Table 3-3. SP-100 Planetary Mission Performance Summary

Mission	Reactor Output Power and Number of Thrusters	Full Power Lifetime, years	Mission Time, years	Target Parameters
Neptune Orbiter	100 kWe/16	5.8	11.2	15.8-RN circular orbit at target
	100 kWe/12	4.4	9.3	*3-RN x 12 days orbit at target
Saturn Ring	100 kWe/16	5.8	8.8	Ring rendezvous, 1.1-RS circular orbit
Rendezvous	100 kWe/12	4.4	6.7	*Ring rendezvous, 1.1-RS circular orbit
Uranus Orbiter	100 kWe/16	5.8	8.8	4.8-RU circular orbit at target
Jupiter Tour	100 kWe/16	4.2	5.1	5.9-RJ circular orbit at target
Pluto/Charon Orbiter	100 kWe/16	5.8	10.9	4-RP circular orbit at target
Mars Surface	100 kWe/12	4.6	5.2	700-km circular orbit at Earth return
Sample Return				500-km circular orbit 300 days at Mars
Asteroid (Vesta) Surface	100 kWe/12	3.7	5.2	700-km circular orbit at Earth return
Sample Return	100 kWe/12	2.1	3.5	*400-km circular orbit at Earth return
Comet (Tempel 2) Nucleus	100 kWe/16	4.9	7.3	700-km circular orbit at Earth return
Sample Return	100 kWe/12	3.3	4.0	*400-km circular orbit at Earth return
Multiple Asteroid (Lumen)	100 kWe/12	4.4	6.4	NEP used to rendezvous at both
Comet (Encke) Rendezvous	100 kWe/12	3.3	5.2	*Asteroid and comet
Venus Orbiter	100 kWe/12	1.7	1.7	300-km circular orbit at target
Mercury Orbiter	100 kWe/12	2.6	2.6	300-km circular orbit at target

*Identifies high thrust chemical (HTC) escape or capture. HTC escape assumes two Shuttle launches and one fully fueled Centaur G' injection stage.

of 16 thrusters or 12 thrusters. Those missions where 16 thrusters are used are constrained to a full power lifetime of 5.8 years and the missions using 12 thrusters are likewise constrained to a full power lifetime of 4.4 years. Note that this latter constraint is not quite met for the Mars sample return mission (4.6 years).

An additional entry in this table is an abbreviated description of the target parameters characterizing each mission. The asterisks prefixing these descriptions indicate that a high thrust chemical (HTC) escape is used for that mission. In these examples, a high thrust chemical capture phase is also employed at the target or the Earth for sample return missions. This high thrust chemical scenario is followed in all of the examples except for that of the Saturn ring rendezvous (SRR) mission where a chemical capture phase is not practical.

The second, Table 3-4, presents a mass summary for these same missions. Except for the Mars sample return mission, the departure mass using a spiral escape mode was constrained to a maximum of about 17,000 kg. In general, the delivered net spacecraft mass varied from around 1000 to 1500 kg, and was defined by a typical engineering and science technology. For some of the examples, estimates of spacecraft mass did not exist and typical payloads were consequently generated. In the case of the comet sample return mission using a high thrust chemical escape and capture at the Earth, the low payload reflects the consequence of performing this mission in the short time of four years. Allowing an additional year or two for the comet sample return mission greatly enhances payload. The spacecraft masses given for the Mars sample return mission correspond to those generated by a recent study at JPL of ballistic Mars sample return missions. This second table also includes a tabulation of the escape and capture modes characterizing each mission.

The last, Table 3-5, presents some mission timelines including spiral escape and capture times, the total heliocentric transfer time, and the length of the different coast phases during each mission. The last two missions, that of a Venus orbiter and a Mercury orbiter, are characterized by continuous thrusting during the mission and do not have any coast phases. Included in this table is a tabulation of the escape and capture energy when these phases are accomplished using a high thrust chemical system. In those examples where an intermediate rendezvous at a planet or small body is required, this table indicates the stay time at that body. Also in the case of the Mars sample return mission, the escape and capture spiral times at Mars are noted.

The following subsections present brief descriptions of each of these missions to complement the data presented in the three tables just described.

3.2.4 Neptune Orbiter Mission

The primary objective of the Neptune orbiter mission is to deliver a 1200- to 1500-kg payload, including a Neptune probe, into a capture orbit at the planet. A relatively loosely bound orbit is specified, and it may be achieved by a short low-thrust capture spiral phase. The performance is

Table 3-4. SP-100 Planetary Mission Mass (kg) Summary

Mission	Initial Mass at Departure	Consumed Propellant	NEP System Mass	Payload Mass	Escape and Capture Modes
Neptune Orbiter	16090	10090	4500	1500	NEP spiral escape and capture
	13870	7570	4000	1500	HTC escape, HTC capture
Saturn Ring Rendezvous	16090	10090	4500	1500	NEP spiral escape and capture
	13070	7570	4000	1500	HTC escape, NEP spiral capture
Uranus Orbiter	16090	10090	4500	1500	NEP spiral escape and capture
Jupiter Tour	13170	7170	4500	1500	NEP spiral escape and capture
Pluto/Charon Orbiter	16090	10090	4500	1500	NEP spiral escape and capture
Mars Surface Sample Return	18770	7970	4000	1000	NEP spiral escape and capture at Earth and Mars (1)
Asteroid (Vesta) Surface	13850	6380	4000	1500	NEP spiral escape and capture (2)
Sample Return	12610	3540	4000	500	HTC escape, HTC capture (2)
Comet (Tempel 2) Nucleus	14450	8450	4500	1000	NEP spiral escape and capture (3)
Sample Return	12170	5680	4000	360	HTC escape and capture at Earth (3)
Multiple Asteroid (Lumen)	14870	7570	4000	2300	NEP spiral escape and rendezvous (4)
Comet (Encke) Rendezvous	13520	5720	4000	2800	HTC escape, NEP rendezvous (4)

Table 3-4. SP-100 Planetary Mission Mass (kg) Summary (contd)

Mission	Initial Mass at Departure	Consumed Propellant	NEP System Mass	Payload Mass	Escape and Capture Modes
Venus Orbiter	9410	2910	4000	2500	NEP spiral escape and capture
Mercury Orbiter	10410	4410	4000	2000	NEP spiral escape and capture

(1) 5800 kg jettisoned at Mars. (2) 2000 kg left at Vesta. (3) 500 kg left at Tempel 2. (4) 1000 kg left at Lumen.

Table 3-5. SP-100 Planetary Mission Timelines

Mission	Escape Spiral, days	Capture Spiral, days	Heliocentric Flight Time, days	Coast Length, days	Intermediate Spiral Capture/ Escape and Wait Time, days Injection C3 for HTC
Neptune Orbiter	404	47	3633 3373	1949 1773	NA *C3 = 2.5 km/s
Saturn Ring Rendezvous	404	645 584	2179 1850	1092 834	NA *C3 = 4.9 km/s
Uranus Orbiter	404	120	2701	1090	NA
Jupiter Tour	329	354	1164	331	NA
Pluto/Charon Orbiter	404	2	3560	1830	NA
Mars Surface Sample Return	431	126	439/393	74/62/64	143/74 spiral at Mars 300-day stay time
Asteroid (Vesta) Surface Sample Return	348	152	741/580 596/565	203/261 143/268	100-day stay time *C3 = 6.5 km/s
Comet (Tempel 2) Nucleus Sample Return	362	152	1233/828 693/665	88/444/165/89 14/101/40	100-day stay time *C3 = 7.8 km/s
Multiple Asteroid (Lumen) Comet (Encke) Rendezvous	373		884/1024 808/1036	215/465 173/410	60-day stay time *C3 = 2.7 km/s
Venus Orbiter	232	172	212	None	NA
Mercury Orbiter	257	57	618	None	NA

*Identifies high thrust chemical (HTC) escape or capture. HTC escape assumes two Shuttle launches and one fully fueled Centaur G' injection stage. NA - not applicable.

calculated assuming a final circular orbit around Neptune at the mean distance of the largest satellite of Neptune, Triton. This final orbit is selected primarily for ease of analysis. Capture into a more eccentric orbit about Neptune with a lower periapsis altitude, but the same semimajor axis, would likely provide higher performance with a larger payload and/or faster flight time. The chemical capture trajectory mode for this mission would place the spacecraft into a somewhat looser orbit around Neptune. This mission also has, by far, the most extended coast phase of any of the missions considered except the Pluto mission. The length of this coast phase of over five years can have important implications on the design of both the power and propulsion subsystems.

3.2.5 Saturn Ring Rendezvous

The primary objective of the Saturn ring rendezvous (SRR) mission is to deliver a 1200- to 1500-kg payload to the inner D-ring of Saturn. Secondary objectives are to deliver a probe to Titan, the largest satellite of Saturn, and to make radar observations of Titan after release of the probe. Although the Saturn mission would appear to be much easier to accomplish than the mission to Neptune, the energy requirements are the same for both missions because of the extended capture spiral maneuvers required for the ring plane observations at Saturn. The ring plane observations are performed as the spacecraft (S/C) continuously thrusts in a retrograde direction and with a small component of thrust directed toward the rings so as keep the spacecraft above (or below) the plane of the ring particles. The spacecraft thus spirals towards the planet on a trajectory which keeps it from about 20 km above the ring plane at the start of the observation period to between 1-2 km at the end of the mission. Ring crossings are also mission options. These distances were calculated assuming a 20-degree thrust vector offset angle to cancel the normal component of gravity acceleration. Although larger values of the thrust offset angle would allow an increase in the distance between the ring plane and spacecraft, it would result in a corresponding increase in capture spiral time and require a higher propellant expenditure for this phase of the mission. Since the propulsion time is constrained for this mission, an increase in propulsion time for the capture phase would have to be balanced by a corresponding decrease in propulsion time for the heliocentric phase. However, this particular heliocentric Saturn trajectory is quite close to the maximum performance limit and a slight decrease in propulsion time would require a much greater increase in heliocentric flight time to accomplish the mission.

3.2.6 Uranus Orbiter

The Uranus orbiter mission would be similar to the Neptune mission. In the Uranus mission example analyzed for this study, a final orbit about Uranus corresponding to the mean distance of the largest satellite, Titania, was selected. The use of this final orbit and a shorter mission time than for the Neptune mission results in an identical energy requirement for these two missions. Delivery of a probe to Uranus, in addition to capture into an equatorial orbit, would likely be more complicated than for the Neptune mission because of the high inclination of the Uranus equatorial plane to the ecliptic plane.

3.2.7 Jupiter Tour

This mission has been treated in a cursory fashion and a detailed description of the encounter phase of the mission has not been performed. Previous studies have indicated that a tour of major Galilean satellites would be an attractive mission. A capture spiral phase is assumed, terminating at the mean distance of Io, the innermost major satellite of Jupiter. Additionally, the mission data presented in the tables has not assumed any additional observation time or probe mass for these four satellites. Other mission designs would likely include observation time at each of the four major satellites and some mass allocation for the probes. These could be accommodated using the present system at the expense of additional propulsion and mission time. An important consideration for the Jupiter tour mission would be the additional radiation imposed on the spacecraft by extended operation time near Jupiter.

3.2.8 Pluto/Charon Orbiter

The objective of this mission is to deliver a 1500-kg payload to the vicinity of Pluto and its satellite, Charon. Because of the weak gravitational field of Pluto, very little energy is required for the capture phase of the mission. Performance was calculated for a final orbit distance of four Pluto radii, although this distance would not be realized at first on an actual mission because of the uncertainties in the physical parameters of Pluto and its satellite. The energy requirements are the same for this mission as for the Neptune mission and the flight time and length of the coast phase are only slightly shorter.

3.2.9 Mars Surface Sample Return

The objective of this mission is the return to Earth of a sample collected from the surface of Mars. The mission scenario includes a spiral capture into a 500-km circular orbit at Mars. Both delivery and return spacecraft system masses are based on those derived in a 1984 Mars Surface Sample Return study, which assumed a net jettison mass of 5800 kg at Mars, a 300-day stay time, and a return spacecraft mass of approximately 1000 kg. Capture at Earth return was accomplished with a capture spiral that placed the spacecraft and NEP subsystem into a 700-km circular orbit. It was assumed that the sample could then be retrieved from this altitude using a space based orbital maneuvering vehicle (OMV) that could be commanded from either the shuttle or a space station.

3.2.10 Asteroid Surface Sample Return

The objective of this mission is similar to that for the Mars Surface Sample Return mission except that the spacecraft mass delivered to the asteroid can be much smaller than that required for Mars. The target asteroid in this study is 4-Vesta and was chosen because of its special scientific interest. In this mission, the NEP subsystem would place the spacecraft into a capture orbit about the body. Little propellant is required for capture into this orbit since even a relatively close orbit would not demand much stay time or propellant consumption. A net jettison mass of 2000 kg and a stay time of 100 days at Vesta was assumed, which should be sufficient to include not only a science station that could be left on the surface but also allowances for low-thrust capture propellant and allowances for a chemical propulsion system that would both land and return the sample to the orbiting spacecraft. This propulsion system would be left at the asteroid although it could be returned to the Earth by the NEP mother ship and used to deliver the sample to the shuttle or space station. A net return spacecraft mass of 1000 kg was assumed for the scenario employing an Earth return low-thrust capture phase, while a net mass of 500 kg was assumed for the mission mode using a hyperbolic return and capture of the spacecraft using a chemical retro propulsion subsystem. This last mode would require an Earth storable propulsion subsystem with a total fueled mass of around 2500 kg at a return hyperbolic excess speed of slightly less than 3 km/sec.

3.2.11 Comet Nucleus Sample Return

Anoter mission analyzed is a comet nucleus sample return to the short periodic comet Tempel 2. The NEP spacecraft would rendezvous with the comet about 50 days prior to comet perihelion and stay with it for 100 to 150 days through perihelion. For returning the sample, a separate autonomous vehicle would be required to carry out the surface science operations and to collect the samples on the surface. A long-life science station would also be left on the surface of the comet for observations through at least one period of the comet around the Sun. The total mission time for the comet sample return mission ranges from four to six years depending on whether high-thrust or low-thrust propulsion is selected for Earth escape and/or Earth return. The relatively small return spacecraft mass resulting from the scenario using a chemical Earth escape and capture mode reflects a desire to perform this mission in the short period of four years. Significantly greater payload capability is possible by allowing an additional year to rendezvous with the comet. Allowing an additional year in the first phase of mission might also allow an earlier arrival at the comet prior to perihelion.

3.2.12 Combined Asteroid and Comet Rendezvous Mission

The objectives of this mission are to rendezvous with and to deliver to these bodies science stations that could be left on their surfaces for long duration scientific observations. This mission was examined to see if it were feasible to rendezvous with two dissimilar small bodies, a comet and an asteroid. Since there are only a few comets that are interesting targets, it is necessary to select the accompanying asteroid to complement the particular comet that is selected. The short periodic comet Encke (3.3 years) was picked as the comet target and a moderate sized asteroid, 141-Lumen with a radius of 58 km, was selected as the asteroid target. In this example, the asteroid is the first target and the comet the second. In most cases, performance is better if the asteroid rendezvous phase is performed first. In this case, it was desirable to rendezvous with the asteroid first since the comet selected has an orbit with a low heliocentric perihelion distance of about 0.35 AU, and there is a strong possibility that the spacecraft could not survive through perihelion. Rendezvous with Encke is at about 50 days before perihelion when the comet is still slightly beyond 1 AU from the Sun. This mission is characterized by a large delivered payload to the comet suggesting that it would be possible to either include other targets in the mission scenario or perform a combined asteroid and comet sample return mission.

3.2.13 Venus Orbiter

The Venus orbiter mission objective is to deliver a large spacecraft to Venus where it would be used to perform extensive atmospheric and surface experiments. The large power available on the NEP spacecraft could be used to perform extensive high resolution mapping of the surface also. This is a relatively low energy mission, and it is possible that such a mission could be performed as well ballistically. Except for the availability of a large source of electric power in Venus orbit, which could be used for various experiments, this mission is not enabled by an NEP subsystem.

3.2.14 Mercury Orbiter

The last mission discussed is an orbiter mission to the innermost planet of the solar system, Mercury. The NEP vehicle would place a large spacecraft of about 2000 kg in a 300-km Mercury orbit. Using Earth-storable propellants, it would be possible to land a science payload of several hundred kilograms on the surface where observations over an extended period of time could be carried out. Delivering this payload to Mercury requires a relatively long heliocentric flight time and results in a heliocentric trajectory with nearly four complete revolutions around the Sun. The primary concern with this mission would be the thermal loads presented to both the spacecraft and the NEP vehicle over a large part of the mission spent close to the Sun.

3.3 SATURN RING RENDEZVOUS MISSION

3.3.1 Why Saturn Ring Rendezvous?

The preceding subsection has briefly described a large set of potential NEP planetary missions. The Saturn Ring Rendezvous Plus Radar (SRRPR) mission was selected from the set in Subsection 3.2 as the mission to focus on for this study. A single mission was selected to focus on in order to allow relatively detailed mission and system designs to be performed. The selected mission was to stress the SP-100 power subsystem as much as, or more than, any other mission in order that a SP-100 power subsystem designed for the most stressing mission would have performance margins for less stressing missions. The SRR mission was selected for the following reasons: (1) the Neptune orbiter mission (more stressing in terms of total mission life) had been studied previously (Reference 3-2), (2) the SRR mission stressed the full power life as much as the Neptune orbiter mission, (3) the SRR mission would likely impose the most severe radiation and particle environments due to its very close approach to Saturn, and (4) the SRR mission is very attractive from the point of view of planetary science objectives.

3.3.2 Saturn Ring Rendezvous Science Objectives and Strawman Payloads

Several options for the nuclear electric propulsion Saturn Ring Rendezvous mission were identified. These options evolve in increasing complexity and performance. They were identified as follows:

- Saturn Ring Rendezvous Plus Radar.
- SRRPR and Titan Probe.
- SRRPR and Titan and Saturn Probes.
- SRRPR and Titan Orbiter (Mapper) and Titan and Saturn Probes.
- SRRPR and Titan Orbiter and Saturn Probe and Titan Semihard Lander.

3.3.2.1 Saturn Ring Rendezvous Plus Radar. This option serves as the building block for the rest of the SRR missions options. It is the standard reference mission which incorporates a 2.5-year Ring Plane Survey, at which time observations are carried out that study Saturn's magnetosphere, atmosphere, and rings. The science objectives of the SRRPR option are as follows:

- Determine the three dimension (3-D) structure and dynamical behavior or Saturn's rings.
- Measure the 3-D structure and dynamical behavior of the magnetosphere.
- Study the chemical composition, physical properties, and dynamical behavior of the atmosphere.
- Characterize the physical and chemical properties of the ring particles.

The instrument classes and expected results are described as follows:

- IMAGING: Ring characteristics, Saturn atmospheric dynamics.
- IR RADIOMETER: Thermal emission as a function of depth in Saturn's atmosphere.
- UV RADIOMETER: Saturnian atmospheric airglow, ring structure to 20-m resolution via stellar occultations.
- MAGNETOMETER, CHARGED PARTICLE DETECTOR, AND PLASMA WAVE ANALYZER: Electrical and magnetic field characteristics and charged particle fluxes of magnetosphere.

- DUST PARTICLE DETECTOR: Determination of particle distribution outside region of rings.

- RADIO SCIENCE: Temperature, pressure profiles in Saturn atmosphere; particle size distribution in rings.

- RADAR: Determine electrical properties of ring particles, measure ring thickness, assist in ring navigation.

A strawman instrument payload for the SRRPR option is shown in Table 3-6.

Table 3-6. Strawman Instrument Payload for SRRPR Option

Instrument	Mass (kg)	Power (W)	Data Rate (bps)
Radar	40	6000	50K
Magnetometer	6	9	300
Plasma Analyzer	12	10	500 (320K)*
Plasma Wave Spectrometer	6	6	240 (650K)*
Energetic Particle Detector	10	10	912
Imaging (Wide Angle)	25	13	95K
Near IR Mapping Spectrometer	24	12	12K
Photopolarimeter	5	4	100
UV Spectrometer	9	8	100 (1K)*
Ultrastable Oscillator	7	4	---
Dust Detector	4	2	24
Imaging (Narrow Angle)	28	13	95K
Total	176	6089	

*These data rates are for peak periods of intensive observations.

3.3.2.2 SRRPR and Titan Probe. This option is the least complicated next to the basic SRRPR mission. Prior tc the Saturn Ring Plane Survey, a probe is delivered to Titan and targeted for a descent mission in excess of one hour. The Titan probe is a Galileo derivative but differs from it in that it carries a preentry science package. This package carries instruments that would measure the characteristics of the outer atmosphere prior to operation of the probe descent instruments. Additionally, the preentry measurements would provide data from altitudes too low for safe orbiter or flyby operation. Upon sensing entry heating, the preentry science package is jettisoned and the descent mission continues. The descent module then continues to characterize the Titan atmosphere, measuring composition and structure, while determining possible energy sources for the chemicals of the organic haze, and imaging the surface below. Upon completion of the probe mission, the spacecraft continues on to its rendezvous with Saturn's rings.

The science objectives for the Titan probe are listed below.

- Determine the structure and chemical composition of the atmosphere.
- Determine the exchange and deposition of energy within the atmosphere.
- Characterize, at least locally, the surface morphology of Titan.

The instrument classes and expected results for the Titan probe are listed below.

Preentry Science:

- ION MASS SPECTROMETER: Composition of the ionosphere.
- NEUTRAL MASS ISOTOPIC SPECTROMETER: Number, density, identification, and ratios of neutral upper atmosphere constituents.
- RETARDING POTENTIAL ANALYZER: Thermal plasma properties and structure of the upper atmosphere.
- ELECTRON TEMPERATURE PROBE: Electron temperatures and electron and ion densities.

Descent Module:

- NEUTRAL MASS SPECTROMETER: Number, density, vertical profile, identification, and isotopic ratios of atmosphere constituents.
- PRESSURE, TEMPERATURE, AND ACCELERATION SENSORS: Mean molecular mass of the atmosphere; upper atmospheric density profile and lower atmosphere pressure,

temperature, and density profiles; horizontal wind velocity, wind sheer, vertical flow, and atmospheric turbulence.

- NEPHELOMETER: Physical structure and location of cloud layers.

- GAS CHROMATOGRAPH: Profiles of trace constituents including the noble gases (neon, argon, krypton), organics (hydrogen cyanide, propane, acetylene, etc.), and carbon monoxide.

- DESCENT IMAGER/RADIOMETER: Vertical distribution of atmospheric constituents such as methane and ammonia and aerosols, by measuring relative light levels at near-infrared and visible wavelengths. Images prior to impact will provide a closeup look at the surface and topography.

A strawman instrument payload for the Titan probe is presented in Table 3-7.

3.3.2.3 SRRPR and Titan Probe and Saturn Probe. This option delivers probes to both Saturn and Titan. Because the mission design is undefined, it is not clear which probe is delivered first. A possible scenario is to first deliver the probe to Saturn on a flyby; then continue on to Titan and deliver the Titan probe. Once the Titan probe mission is completed, the spacecraft would then proceed with the ring rendezvous mission. Possible targeting for the Saturn probe would be a below-the-rings pass entering at mid-southern latitudes. Because of less severe entry heating conditions than at Jupiter, the Saturn probe is able to use a thinner heat shield. This heat shield protects instruments and electronics sensitive to entry heating while pressure, temperature, and acceleration measurements provide information to reconstruct the upper atmosphere. After slowing, a parachute separates the descent module from the heat shield, allowing the descent module to sample the atmosphere.

The science objectives, instrument classes and expected results, and a strawman instrument payload (Table 3-8) for the Saturn probe are presented below.

- Those objectives for SRRPR and Titan Probe.

- Determine the chemical composition and physical properties of the atmosphere of Saturn.

3.3.2.4 Instrument Class and Expected Results - (Saturn Probe).

- NEUTRAL MASS SPECTROMETER: Number, density, vertical profile, identification, and isotopic ratios of atmospheric constituents.

Table 3-7. Titan Probe Strawman Instrument Payload

Instrument	Mass (kg)	Power (W)	Data Rate (bps)
<u>Preentry</u>			
Ion Mass Spectrometer	2.9	1.5	6
Neutral Mass Isotopic Spectrometer	4.3	13	6
Retarding Potential Analyzer	2.7	2.8	18
Electron Temperature Probe	2.0	4.0	6
Power/Telemetry Interface + Structure, Cable, Harness	12	1	---
<u>Descent Module</u>			
Neutral Mass Spectrometer	12	29.5	32
Atmospheric Structure Instrument	4	6	18
Nephelometer	5	14	10
Gas Chromatograph	4	20	20
Descent Imager/Radiometer	3	7	16
Subtotal	51.9	98.8	
Galileo Deceleration Module (includes heat shield)	213		
Descent Module (includes Science)	141		
Probe Total	354*		

*Probe total may be significantly reduced if a thinner heat shield is used due to less severe heating from Titan atmosphere.

Table 3-8. Saturn Probe Strawman Instrument Payload

Instrument	Mass (kg)	Power (W)	Data Rate (bps)
Neutral Mass Spectometer	12	29.5	32
Atmospheric Structure Instrument	4	6	18
Nephelometer	5	14	10
Helium Abundance Detector	1.5	1	4
Lightning and Radiation Detector	2	2	8
Net Flux Radiometer	3	10	16
Gas Chromatograph	4	20	20
Subtotal	31.5	82.5	
Deceleration Module	139		
Descent Module (Science included)	121		
Probe Total	260		

- PRESSURE, TEMPERATURE, AND ACCELERATION SENSORS: Mean molecular mass of the atmosphere; upper atmospheric density profile and lower atmosphere pressure, temperature, and density profiles; horizontal wind velocity, wind shear, vertical flow, and atmospheric turbulence.

- NEPHELOMETER: Physical structure and location of cloud layers.

- HELIUM ABUNDANCE DETECTOR: Accurate hydrogen/helium abundance ratio in the atmosphere.

- LIGHTNING AND RADIATION DETECTOR: Verification of the presence of lightning; scale size of cloud turbulence.

- NET FLUX RADIOMETER: Location of cloud layer; variation in the mixing ratios of atmospheric constituents, energy transport, and deposition in the atmosphere.

- GAS CHROMATOGRAPH (substitution for helium abundance detector and lightning radiation detector): Profiles of trace constituents including noble gases (neon, argon, krypton), organic and inorganic molecules, sulfur compounds, and water.

3.3.2.5 SRRPR and Titan Probe and Saturn Probe and Titan Orbiter. This option would provide mapping of Titan's surface along with delivering two probes and rendezvousing with the ring plane. The radar would be capable of imaging Titan's surface from 2000 m to a resolution of 1 km. The operational capability of the radar would allow large portions of the satellite surface (possibly 20%) to be imaged per orbit. As on the basic SRRPR option, the radar would operate during the ring plane survey determining particle sizes and distribution. For the orbiting option, trades exist for radar mass, size, power, and resolution. Lower operational altitudes would translate to reduced size and mass, increased resolution, or lower power. If orbit capabilities (spacecraft) are precluded, this radar version could also operate in a flyby mode imaging significant portions of the surface.

The science objectives and instrument classes and expected results for the Titan orbiter are listed below. The strawman instrument payload would be identical to the combination of Tables 3-6, 3-7, and 3-8.

3.3.2.6 Science Objectives - (Titan Orbiter).

- Determine the composition, morphology, and physical state of the surface.
- Determine the geologic history of Titan.
- Study the time variability of Titan's clouds/hazes.

3.3.2.7 Instrument Class and Expected Results - (Titan Orbiter).

- RADAR: Surface topography and morphology.
- RADIO SCIENCE: Temperature and pressure profiles in planetary atmospheres from radio occultations.
- MAGNETOMETER AND CHARGED PARTICLE DETECTOR: Electrical and magnetic field characteristics and charged particle fluxes of (possible) magnetosphere.
- DUST PARTICLE DETECTOR: Determination of particle distribution.

3.3.2.8 SRRPR and Titan Orbiter and Saturn Probe and Titan Semihard Lander. This option includes all of the previous options with one significant change; the Titan probe is reconfigured to a semihard lander. The most recent information about Titan suggests that the surface may be liquid ethane. If this is the case, the lander may need to be a "floater" as well. The semihard lander can be configured with three science packages:

(1) Landed Science Package.

(2) Landed Science and Descent Science.

(3) Landed Science and Descent Science and Preentry Science.

Principal to this design is the inclusion of a small retromotor for terminal braking (semihard impact). It would be possible to carry out this mission without the mapping (orbiter) option; but for maximum science return the semihard lander would be coupled with the orbiting spacecraft option. A possible option on the landing system might include a penetrator/support structure (i.e., pogo-stick/penetrator landing structure).

The science objectives for the Titan semihard lander are listed below. Table 3-9 presents a strawman instrument payload.

3.3.2.9 Science Objectives - (Titan Semihard Lander).

- Determine structure and chemical composition of the atmosphere.
- Determine the exchange and deposition of energy within the atmosphere.
- Characterize, at least locally, the surface morphology of Titan.

Table 3-9. Titan Semihard Lander Strawman Instrument Payload

Instrument	Mass (kg)	Power (W)	Data Rate (bps)
Titan Probe Payload	51.9	98.8	TBD
Magnetometer	0.6	0.3	TBD
Seismometer	0.9	0.2	TBD
α-Proton Backscatter/X-Ray Fluorescence Spectrometer	2.0	1.2	TBD
γ-Ray Spectrometer	3.0	2.0	TBD
Subtotal	58.4	102.5	

- Surface composition, morphology, and features.
- Interior structure.

3.3.3 Saturn Ring Rendezvous Mission Description

The information in this subsection is a condensation of that presented in Reference 3-3 and describes only the basic Saturn Ring Rendezvous (SRR) mission. NEP missions to Saturn occur roughly on yearly intervals. The NEP SRR spacecraft is launched by the space shuttle into a nominal 280 km, 28.5° circular low Earth orbit. A small chemical propulsion system (see Subsection 4.6) then boosts the system to a 700 km, 28.5° circular orbit. A 700-km orbit has an orbital lifetime of about 300 years (for a ballistic coefficient consistent with this type of spacecraft integrated with a SP-100 power subsystem). The SP-100 power subsystem and the propulsion subsystem are started and the spacecraft begins the Earth escape phase of the mission. At this time, the spacecraft has a mass of about 17,000 kg, of which approximately 10,000 kg is mercury propellant. The spiral escape from Earth lasts about 425 days with the spacecraft making about 1700 orbits of the Earth. Figure 3-4 presents a reference SRR heliocentric trajectory. The first part of the heliocentric powered flight phase continues through to about 660 days after launch when a short 225-day coast is employed. Continuous thrusting again begins at 885 days (2.4 years) and lasts through to 1392 days (3.8 years) after launch, at which time the spacecraft begins a long coast lasting until 2842 days (7.8 years) after launch. At this time, the powered flight begins again in order to reduce heliocentric orbit energy.

Saturn rendezvous is attained at 3075 days (8.4 years) after launch at which time the Saturn capture spiral phase is initiated. Passage of Titan orbital radius occurs at 3175 days (8.7 years) after launch. Thrusting parallel to the ring plane continues until a 3.0-R_S Saturn radius is achieved at 3502 days (9.6 years) just outside the G-ring when the ring spiral phase begins. At this time, the thrust vector is offset creating a small component of the thrust vector normal to the rings in order to boost the spacecraft continuously out of the ring plane in a non-Keplerian, minor circle orbit. For the purposes of this mission description, a 20° offset angle with respect to the ring plane is assumed. This offset angle places the NEP spacecraft about 18 km above the plane of the G-ring and reduces the thrust component available for spiraling inbound toward Saturn by about 6%. Inbound spiraling continues with a constant offset angle and a continually reducing altitude above the ring plane until 1.1 R_S is achieved at a final altitude above the rings of about 1 km at 3810 days (10.4 years) after launch from Earth. Figure 3-5 is a schematic drawing showing the near Saturn Ring spiral phase. The top of the figure shows the major Saturn rings and the lower part illustrates the spacecraft orbit trace from a view in the ring plane.

Using a constant 20° thrust offset angle, the time required to spiral from the G-ring to the inner edge of the D-ring at 1.1 R_S is 308 days. Figure 3-6 shows the average rate of relative motion between the spacecraft and the particles directly below the spacecraft as a function of Saturn distance. This relative speed is also the average rate of radius reduction during the spiral phase. Figures 3-7 and 3-8 show the relative

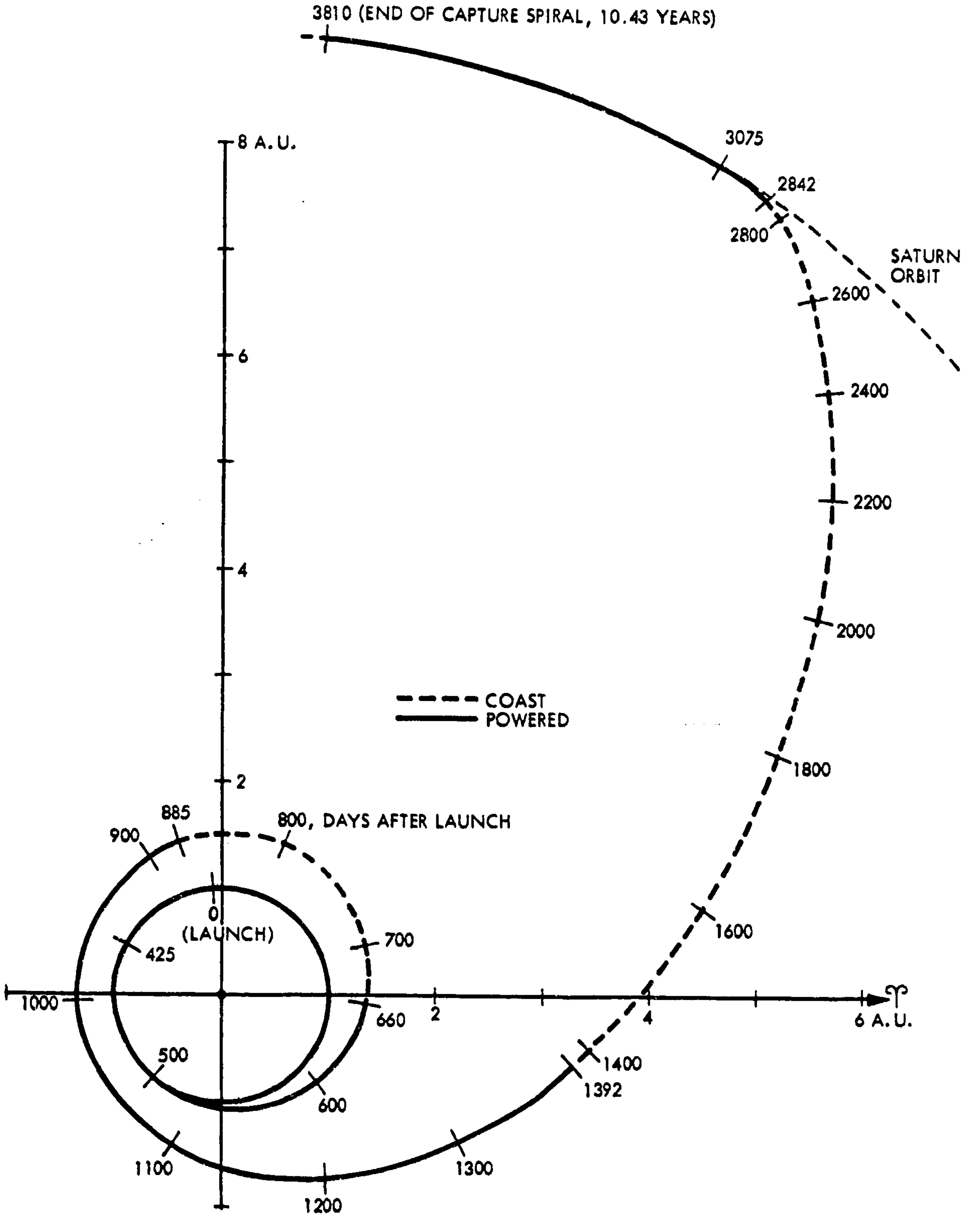

Figure 3-4. Reference Heliocentric Saturn Ring Rendezvous Mission Trajectory

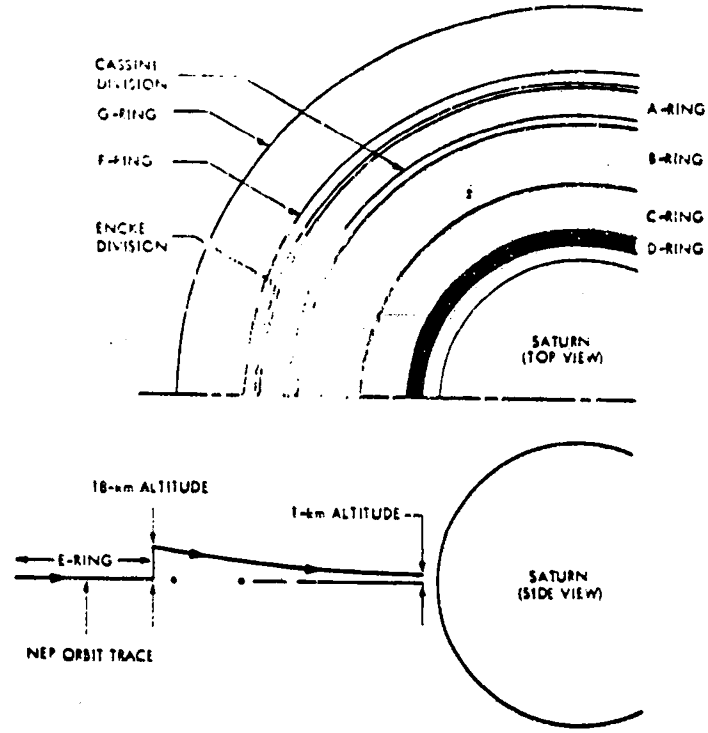

Figure 3-5. Nominal Geometry (Not to Scale) of the Ring Survey Mission Phase

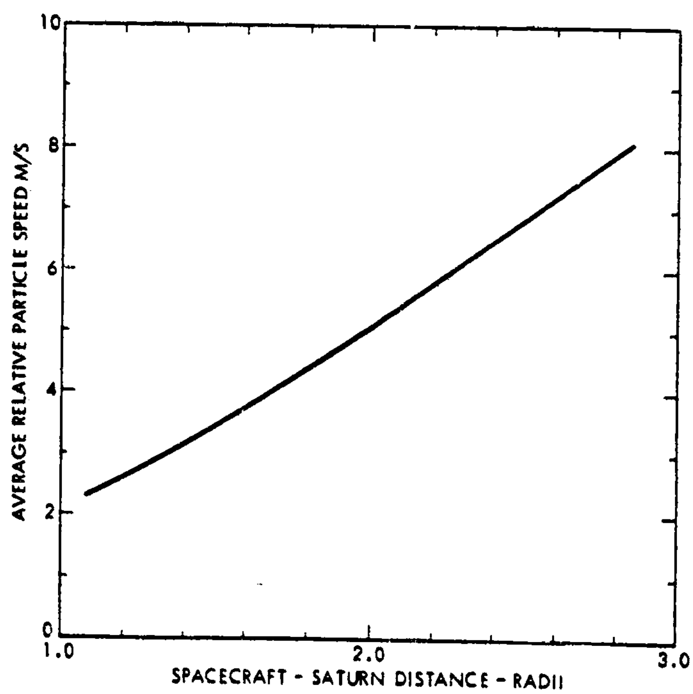

Figure 3-6. Average Relative Spacecraft-Particle Speed Versus Distance from Saturn

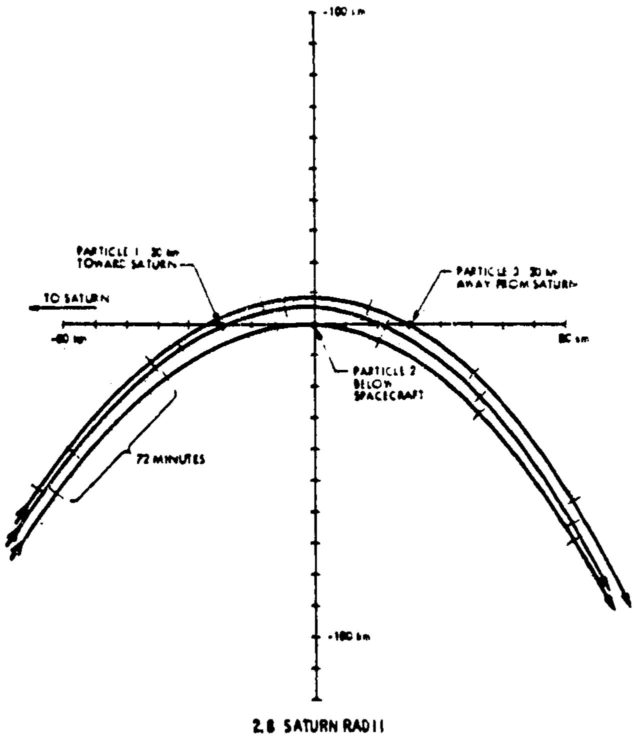

Figure 3-7. Relative Spacecraft-Ring Particle Motion (Rotating Coordinate Frame) at 2.8-Saturn Radii

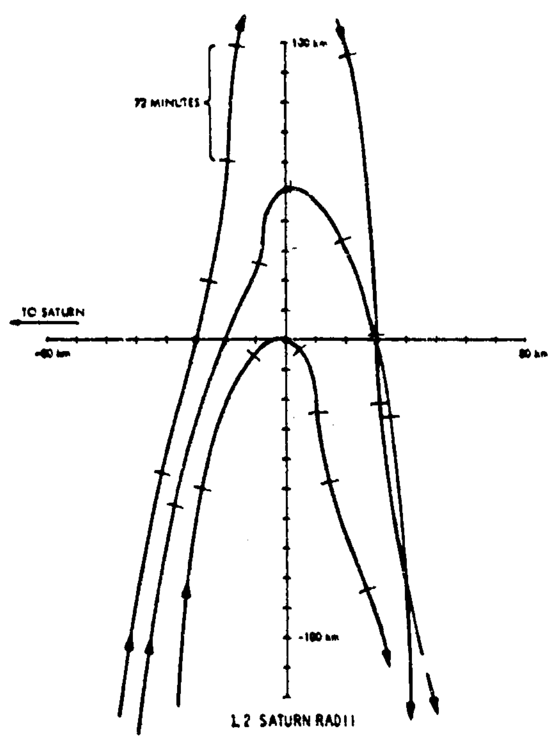

Figure 3-8. Relative Spacecraft-Ring Particle Motion at 1.2-Saturn Radii

motion of the spacecraft between three particular ring particles for two Saturn radii: 2.8 and 1.2. In these figures, time-coincident positions of the three ring particles, shown as points on the horizontal axes, are 30-km toward Saturn, below the spacecraft, and 30-km away from Saturn. The paths of the particles are integrated forward and backward in time. The deviation from near parabolic motion seen in Figure 3-8 at 1.2 R_S occurs because the spacecraft orbit is slightly elliptical due to the assumed fixed, nonoptimal thrust pointing direction, namely exactly perpendicular to the radius vector. At 1.2 R_S, this effect is more pronounced because the motion shown occurs roughly over one orbital period.

3.3.4 Saturn Ring Rendezvous Timelines

From the five mission options presented in Subsection 3.3.2, four were selected as the mission options for this study. These four are listed below and do not include a Saturn probe or Titan semihard lander. The four mission options (cases) are listed in order of increasing complexity.

(1) Basic Saturn Ring Rendezvous Plus Radar (SRRPR).

(2) SRRPR and Titan Probe.

(3) SRRPR and Titan Orbiter and radar mapping of the entire surface of Titan from a 2000-km polar orbit.

(4) SRRPR, Titan Probe, and Titan surface mapping.

The timelines are characterized by the inclusion or noninclusion of the Titan probe, which would be dropped at Titan during the capture spiral at Saturn, and by the inclusion or noninclusion of the Titan capture and orbit phase for the radar mapping option.

As discussed in Subsection 4.2, the parameters adopted for the NEP propulsion system for this mission are (1) an SP-100 power level of 100 kWe, (2) a propulsion system specific impulse of 5300 seconds, (3) a thrust level of 2.84 newtons, and (4) a total propulsion system mass of 5250 kg. A net spacecraft mass of 1550 kg and a Titan probe mass of 350 kg are assumed for this mission. A nominal low-thrust propellant loading of 10,086 kg is used for the mission options, which do not include the Titan mapper phase. Those options, which include the Titan mapper phase, require approximately four to eight percent additional propellant loading to accomplish the mission. The Earth escape spiral starts from a circular orbit altitude of 700 km while the capture spiral orbit at Saturn ends at approximately 6000 km above Saturn at the edge of the inner D-ring.

The information in this subsection is presented in the form of five tables of data and three figures. Table 3-10 presents trajectory event times for the four case mission options being considered. The tabulated times for the end of the Earth escape spiral and the start of the Saturn capture spiral are fictitious and represent the effective spiral times for performance calculations. They are not the actual times of escape or capture. Spiral capture times at Titan, for the two options involving a Titan mapper phase, were calculated assuming a final circular orbit of 2000-km altitude at Titan.

Table 3-10. Event Time/Mass Summary for Saturn Ring Rendezvous Mission

Event Times (days)	Case 1	Case 2	Case 3	Case 4
Launch	0	0	0	0
End of escape spiral	425	434	435	453
Start of first coast	660	711	712	738
End of first coast	885	994	994	995
Start of second coast	1392	1463	1465	1498
End of second coast	2842	3156	3156	3155
Saturn arrival and start of capture spiral phase	3075	3372	3372	3378
Titan encounter (1)	3176	3476	3475	3488
End of Titan capture spiral and start of Mapper phase			3515	3528
End of Mapper phase and start of Titan escape spiral (2)			3575	3588
End of Titan escape spiral			3614	3627
Start of Ring Rendezvous (3)	3502	3804	3942	3955
End of mission (4)	3810	4111	4250	4262
Total mission time (years)	10.43	11.26	11.64	11.67
Mass Summary				
Initial mass (700-km orbit)	16886	17236	17288	17957
Consumed propellant	10086	10086	10488	10807
Titan probe mass		350		350
S/C dry mass	6800	6800	6800	6800

Case 1: Baseline Ring Rendezvous mission only.
Case 2: Baseline mission + Titan probe.
Case 3: Baseline mission + 60 day Titan mapper phase.
Case 4: Baseline mission + Titan probe + 60 day Titan mapper phase.

Notes:
(1) Probe release and/or start of Titan capture phase.
(2) Net stay time at Titan is four orbital periods or 60 days.
(3) Ring Rendezvous phase starts at 3.0 Rs.
(4) End of Ring Rendezvous phase and mission is at 1.1 Rs.

Additional note: Patch times for spiral escape and capture phases are "effective" times for performance calculations. Actual times of zero relative central body energy are slightly different. Unless otherwise noted, all times are in days.

This orbit altitude could be lowered with a slight increase in spiral time and a slightly higher low-thrust propellant loading. A total stay time of 60 days in orbit at Titan for the mapper phase is used, which represents about four revolutions of the satellite about Saturn. This time would probably be sufficient for the science observations; however, it can be increased or decreased with a corresponding change in total mission time.

A short mass summary is also included in Table 3-10. The final spacecraft dry mass represents the total of both the propulsion system mass and net spacecraft mass. The first two cases presented in Table 3-10 were calculated using the constrained propellant loading mass used in previous studies. Since this mission is very near the maximum performance limit for the assumed propulsion system parameters, it was necessary to augment the mission in some way in order to accomplish the additional Titan mapper phase with its additional propulsion requirements. This was done by increasing the low-thrust propellant loading slightly.

Table 3-11 presents tabular data for the Earth escape spiral phase. This data includes distance from Earth or Saturn in planet radii, time from start of mission, revolutions about the Earth, and total spacecraft mass. The data in this table was generated for the baseline mission (SRRPR) as was that for the capture spiral phase at Saturn shown in Tables 3-12 to 3-14. Table 3-12 presents capture spiral data at Saturn from approach to Titan orbit. In these tables, both time and orbit revolutions are referenced from the end of the mission. This time thus represents time-to-go until the end of mission. Table 3-13 presents data from Titan orbit to the beginning of the Ring Rendezvous Phase at 3 Saturn radii and Table 3-14 presents the spiral data during the Ring Rendezvous phase from 3 to 1.1 Saturn radii. The altitude above or below the ring plane is also shown in Table 3-14 for a constant thrust offset angle of 20 degrees. This altitude ranges from 18 km at the start of the Ring Rendezvous phase to 1 km at the end of the mission. Increasing this altitude by increasing the thrust offset angle would have a serious consequence on mission performance because of the resulting increase in spiral capture time. Note that Tables 3-13 and 3-14 are the same for all four mission options since the spacecraft parameters are the same for all options following Titan.

A plot of spacecraft distance from the Earth during the escape spiral phase is shown in Figure 3-9 as a function of time from the start of the mission. A similar plot for the capture phase at Saturn is shown in Figure 3-10. A graphical presentation of the trajectory timelines is presented in Figure 3-11 for the four mission options indicating both propulsion and event times. The long second coast phase during the heliocentric phase of the trajectory has been truncated for clarity in this figure. This Saturn Ring Rendezvous mission is interesting in that the majority of propulsion occurs during the escape and capture phases of the mission rather than during the heliocentric portion as is the case for most electric propulsion missions.

Table 3-11. Earth Escape Spiral

F= 2.84239 N IS= 5300 sec MO= 16886 kg

RADI	TIME	REV	ALT	MASS
1.110	.00	.00	.000	16886.0
1.201	20.00	275.04	.000	16791.5
1.305	40.00	518.57	.000	16697.0
1.424	60.00	732.95	.000	16602.5
1.561	80.00	920.45	.000	16508.0
1.719	100.00	1083.29	.000	16413.5
1.904	120.00	1223.60	.000	16319.0
2.122	140.00	1343.46	.000	16224.5
2.381	160.00	1444.85	.000	16130.0
2.692	180.00	1529.69	.000	16035.5
3.071	200.00	1599.81	.000	15941.0
3.539	220.00	1656.94	.000	15846.5
4.128	240.00	1702.73	.000	15752.0
4.881	260.00	1738.74	.000	15657.5
5.867	280.00	1766.43	.000	15563.0
7.195	300.00	1787.15	.000	15468.5
9.044	320.00	1802.16	.000	15374.0
11.729	340.00	1812.58	.000	15279.5
15.846	360.00	1819.45	.000	15185.0
22.630	380.00	1823.67	.000	15090.5
34.967	400.00	1826.02	.000	14996.0
60.362	420.00	1827.17	.000	14901.5
117.089	440.00	1827.64	.000	14807.5
234.157	460.00	1827.82	.000	14712.5
434.445	480.00	1827.89	.000	14618.0
725.915	500.00	1827.93	.000	14523.5
1109.923	520.00	1827.94	.000	14429.0
1586.786	540.00	1827.95	.000	14334.5
2156.866	560.00	1827.56	.000	14240.0

Table 3-12. Saturn Spiral Capture into Titan Orbit

F= 2.84239 N   IS= 5300 sec   MO= 6800 kg

RADI	TIME	REV	ALT	MASS
20.370	635.427	962.77	.000	9802.4
20.755	640.000	963.06	.000	9824.0
23.732	650.000	963.60	.000	9871.2
25.164	660.000	964.09	.000	9918.5
28.947	670.000	964.50	.000	9965.7
31.344	680.000	964.84	.000	10013.0
34.849	690.000	965.15	.000	10060.2
40.084	700.000	965.40	.000	10107.5
45.318	710.000	965.61	.000	10154.7
50.705	720.000	965.78	.000	10202.0
57.245	730.000	965.93	.000	10249.2
65.550	740.000	966.05	.000	10296.5
75.779	750.000	966.15	.000	10343.7
87.973	760.000	966.23	.000	10391.0
102.230	770.000	966.30	.000	10438.2
118.721	780.000	966.35	.000	10485.5
137.649	790.000	966.39	.000	10532.7
159.214	800.000	966.43	.000	10580.0
183.595	810.000	966.46	.000	10627.2
210.935	820.000	966.48	.000	10674.5
241.344	830.000	966.50	.000	10721.7
274.899	840.000	966.51	.000	10769.0
311.650	850.000	966.53	.000	10816.2
351.628	860.000	966.54	.000	10863.5
394.847	870.000	966.55	.000	10910.7
441.311	880.000	966.56	.000	10958.0
491.018	890.000	966.56	.000	11005.2
543.958	900.000	966.57	.000	11052.5
600.121	910.000	966.57	.000	11099.7
659.494	920.000	966.58	.000	11147.0

*Note: Times and revolutions measured from end-of-mission.

Table 3-13. Saturn Spiral Capture from Titan Orbit
to Start of Ring Phase at 3 Rs

F= 2.84239 N IS= 5300 sec MO= 6800. kg

RADI	TIME	REV	ALT	MASS
3.000	307.430	823.19	.000	8252.6
3.500	343.990	859.42	.000	8425.4
4.000	374.017	883.39	.000	8567.2
4.500	399.271	900.07	.000	8686.6
5.000	420.910	912.16	.000	8788.8
5.500	439.730	921.20	.000	8877.7
6.000	456.301	928.14	.000	8956.0
6.500	471.042	933.59	.000	9025.7
7.000	484.271	937.95	.000	9088.2
7.500	496.232	941.48	.000	9144.7
8.000	507.118	944.39	.000	9196.1
8.500	517.081	946.82	.000	9243.2
9.000	526.247	948.86	.000	9286.5
9.500	534.717	950.60	.000	9326.5
10.000	542.576	952.09	.000	9363.7
10.500	549.895	953.37	.000	9398.2
11.000	556.734	954.49	.000	9430.6
11.500	563.142	955.47	.000	9460.8
12.000	569.165	956.33	.000	9489.3
12.500	574.839	957.10	.000	9516.1
13.000	580.197	957.77	.000	9541.4
13.500	585.267	958.38	.000	9565.4
14.000	590.076	958.92	.000	9588.1
14.500	594.643	959.41	.000	9609.7
15.000	598.990	959.86	.000	9630.2
15.500	603.134	960.26	.000	9649.8
16.000	607.090	960.62	.000	9668.5
16.500	610.871	960.95	.000	9686.4
17.000	614.491	961.26	.000	9703.5
17.500	617.961	961.54	.000	9719.9
18.000	621.290	961.80	.000	9735.6
18.500	624.489	962.03	.000	9750.7
19.000	627.566	962.25	.000	9765.2
19.500	630.527	962.45	.000	9779.2
20.000	633.381	962.64	.000	9792.7
20.370	635.427	962.77	.000	9802.4

Table 3-14. Saturn Spiral Capture from Start of Ring Phase to End of Mission at 1.1 Rs

F= 2.84239 N   IS= 5300 sec   MO= 6800 kg
Thrust Offset Angle = 20 deg

RADI	TIME	REV	ALT	MASS
1.100	.000	.00	1.084	6800.0
1.150	15.610	75.42	1.225	6873.8
1.200	30.387	142.31	1.378	6943.6
1.250	44.406	201.92	1.543	7009.8
1.300	57.729	255.27	1.720	7072.8
1.350	70.416	303.22	1.910	7132.7
1.400	82.515	346.48	2.113	7189.9
1.450	94.073	385.64	2.330	7244.5
1.500	105.129	421.21	2.561	7296.7
1.550	115.720	453.63	2.807	7346.8
1.600	125.877	483.25	3.067	7394.8
1.650	135.632	510.39	3.343	7440.9
1.700	145.009	535.32	3.634	7485.2
1.750	154.035	558.28	3.942	7527.8
1.800	162.729	579.47	4.266	7568.9
1.850	171.113	599.07	4.608	7608.5
1.900	179.206	617.23	4.967	7646.7
1.950	187.023	634.10	5.343	7683.7
2.000	194.581	649.80	5.738	7719.4
2.050	201.894	664.42	6.152	7753.9
2.100	208.976	678.07	6.585	7787.4
2.150	215.837	690.84	7.037	7819.8
2.200	222.491	702.79	7.509	7851.3
2.250	228.947	714.00	8.002	7881.8
2.300	235.215	724.53	8.515	7911.4
2.350	241.305	734.43	9.050	7940.2
2.400	247.224	743.75	9.606	7968.1
2.450	252.981	752.53	10.184	7995.3
2.500	258.583	760.83	10.785	8021.8
2.550	264.037	768.66	11.408	8047.6
2.600	269.351	776.07	12.055	8072.7
2.650	274.528	783.08	12.726	8097.1
2.700	279.577	789.73	13.420	8121.0
2.750	284.501	796.04	14.139	8144.3
2.800	289.306	802.03	14.883	8167.0
2.850	293.998	807.73	15.652	8189.1
2.900	298.579	813.14	16.447	8210.8
2.950	303.055	818.30	17.268	8231.9
3.000	307.430	823.21	18.115	8252.6

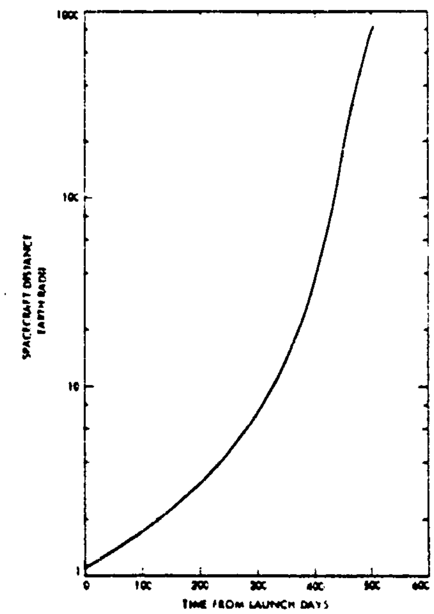

Figure 3-9. Distance Versus Time of the Earth Escape Portion of the Nominal Saturn Ring Rendezvous Mission

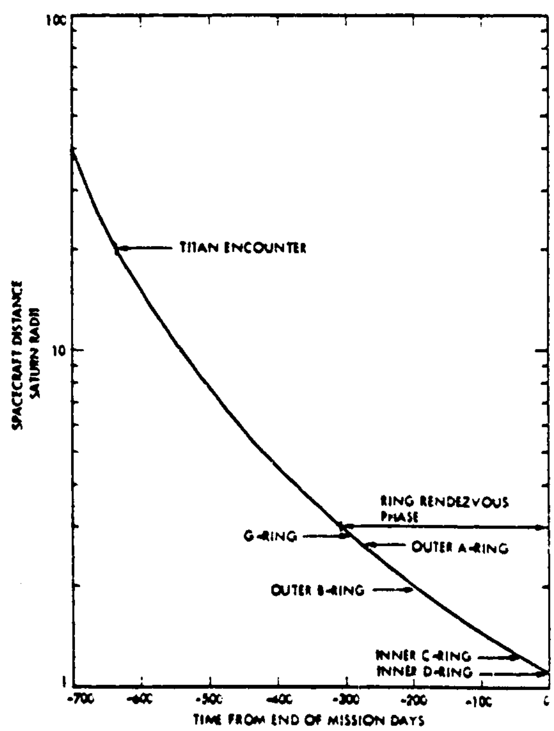

Figure 3-10. Distance Versus Time of the Saturn Capture Spiral of the Nominal Saturn Ring Rendezvous Mission

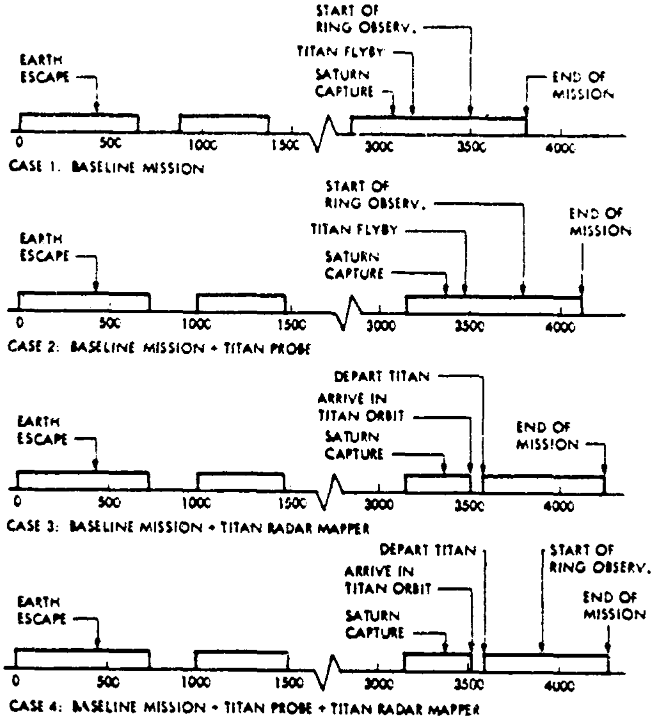

Figure 3-11. Saturn Ring Rendezvous Timelines Showing the Full Power/Propulsion Periods

SECTION 4

SYSTEM DESIGN

4.1 SP-100

As discussed in the Introduction (Section 1), it was not the purpose of this study to design the SP-100 power subsystem. This study merely accepted the SP-100 as defined by the System Definition element of the SP-100 Project and proceeded to integrate it with the Saturn Ring Rendezvous mission and spacecraft. This section summarizes the SP-100 as it was defined for purposes of this study.

Four SP-100 power subsystem concepts are being studied. Table 4-1 presents a summary of the major characteristics of these four concepts. Each power subsystem concept has been designed to meet the following major requirements.

(1) End of Life Power - 100 kWe.

(2) Mass - 3000 kg or less.

(3) Launch Vehicle - NASA space shuttle.

(4) Launch Configuration - No larger than one-third of the shuttle payload bay length.

(5) Full Power Life - 7 years.

(6) Total System Life - 10 years.

(7) Seven-Year Radiation Dose at 25 m - 5×10^5 Rad (Si) and 1×10^{13} neutrons/cm^2.

4.2 ELECTRIC PROPULSION

For this study, ion thrusters using mercury propellant were assumed. This assumption was made based upon a long history of using mercury ion thrusters in studies of this kind and the lack of any serious alternative to mercury ion thrusters for the power level and probable time period of the Saturn Ring Rendezvous mission. The basic starting point for this subsection was a 100-kWe constant power SP-100 power subsystem. Reference 4-1 reviewed SP-100 nuclear electric propulsion outer planet missions and found that for a 100-kWe SP-100 power subsystem and mercury ion thrusters, a specific impulse of 5300 sec was appropriate.

Using the design point of 100 kWe and 5300 sec, mercury ion thruster propulsion subsystem preliminary designs were developed based upon past work (Reference 4-2) and a propulsion subsystem model, which is based upon previous models and point designs (References 4-2, 4-3, and 4-4). During 1984, the ion thruster staff of the NASA Lewis Research Center (LeRC) supplied projections for the ion thruster technology parameters that are necessary

Table 4-1. SP-100 Power Subsystem Concept Characteristics

	Thermionic	Thermoelectric	Stirling	Brayton
Reactor				
Spectrum	Fast	Fast	Fast	Fast
Fuel Element	Pin	Pin	Pin	Pin
Coolant	Liquid Metal	Liquid Metal	Liquid Metal	Liquid Metal
Fuel	UO_2	UN	UO_2	UN
Emitter Temperature	1600-1900 K	N.A.*	N.A.	N.A.
Outlet Temperature	900-1100 K	1300-1500 K	900-1100 K	1300-1500 K
Structural and Cladding Material	Refractory Metal	Refractory Metal	Refractory Metal or Stainless Steel	Refractory Metal
Power Conversion				
Method	Thermionic	Thermoelectric	Free Piston Stirling and Linear Alternator	Brayton Engine Direct Coupled Alternator
Location	In Core	Out of Core	Out of Core	Out of Core
Heat Transport				
Reactor to Converter				
Working Fluid	N.A.	Liquid Metal	Liquid Metal	Liquid Metal
Pipe Material		Refractory Metal	Refractory Metal or Stainless Steel	Refractory Metal
Converter to Radiator				
Working Fluid	Liquid Metal	Liquid Metal	Liquid Metal	Inert Gas or Liquid Metal
Radiator				
Temperature	800-1100 K	600-900 K	400-800 K	400-800 K
Deployment Required	No	Yes	Yes	Yes

*N.A. = Not applicable.

inputs to the propulsion subsystem model referred to in the preceding sentence. These projections were for two technology levels: (1) a relatively near term projection for 30-cm thruster technology and (2) an advanced projection for 50-cm thruster technology. One of the most important projections that was included for both of these technology levels was a projection for the ion thruster power processor mass as a function of power. Based upon the LeRC projections, the lifetimes for the 30- and 50-cm thrusters were selected to be 1×10^5 Ah and 2.8×10^5 Ah, respectively. This lower lifetime is somewhat higher than the LeRC near term projected lifetime; but the longer lifetime is significantly below the LeRC projected advanced technology lifetime. Using the technology projections and subsystem models, two ion thruster propulsion subsystem preliminary designs were developed. A summary of these preliminary designs is presented in Table 4-2. The propellant tank mass is an allocation rather than a preliminary design. The power processor radiation area is to be used in the configuration and mass properties (Subsection 4.4). The thermal control for the subsystem consists of single sided radiators, heat pipes, and multilayer insulation. The radiators are sized to dissipate the power processor waste power and maintain the power processor critical baseplate temperature at 50°C. Note that both subsystems can operate at full power and leave approximately 2 kWe for use elsewhere in the spacecraft.

The electrical interface is the primary link between the electric propulsion subsystem and the SP-100 power subsystem. This interface was briefly studied in order to uncover any specific requirements that needed to be placed upon the SP-100 power subsystem in order that it could be successfully integrated with the electric propulsion subsystem. No intrinsic or fundamental conflicts were found and no new or modified SP-100 power subsystem requirements were identified. The basic reason for this compatibility is that the ion engine power processor is very complex and has incorporated so many features into its current design philosophy that it is able to accept very "raw" power. The ion engine power processor design philosophy has had to accommodate power from a solar array whose power varies with time and whose voltage may vary over a factor of two from the start to the end of a mission. The ion engine power processor is made up of many individual DC-AC-DC power supplies (5 to 12 depending upon the design approach) each with its own control requirements and interface with the power distribution system. Since these power supplies already have significant control and power conditioning capability, these functions within the SP-100 power subsystem can be kept at the minimum that is required for transient and safety considerations. Again, because each power supply will do most of its own power conditioning, the voltage regulation on the power distribution system can be as large as 10% without concern. A straight forward shunt regulator in the SP-100 power subsystem should be completely compatible with a group of ion engine power processors. The same designer awareness and protection philosophy required to handle transient phenomena between thrusters and power supplies has been extended to the power distribution interface where the effects of periodically switching the several modular units on- and off-line have been considered.

Because no attempt was made to integrate a particular ion engine power processor design with the SP-100 power subsystem, the assertion that there are no major interface problems must be taken as a default statement.

Table 4-2. Saturn Ring Rendezvous Ion Thruster Propulsion Subsystems (5300 sec)

Thruster Size	30 cm	50 cm
Thrust per Thruster	0.72 N	1.49 N
Maximum Number of Operating Thrusters	4	2
Total Number of Thrusters	16	6
Maximum Number of Operating Power Processors	4	2
Total Number of Power Processors	8	4
Thruster Beam Current	6.36 A	13.1 A
Thruster Life	15,700 hr	21,400 hr
Required Mission Life	51,380 hr	51,380 hr
Excess Subsystem Life	22%	25%
Power Processor Input Power	24.3 kWe	49.1 kWe
Thrust Module Masses	1320 kg	738 kg
Thrusters	192	174
Thruster Support Structure and Actuator	144	132
Power Processors	592	176
Power Processor Radiators and Thermal Control	360	244
Miscellaneous	32	12
Interface Module Masses	725 kg	503 kg
Power Processing	524	390
Harness	48	18
Thermal Control	16	16
Structure	97	59
Thrust Subsystem Controller	8	8
Miscellaneous	32	12
Propellant Tank	100 kg	100 kg
Power Processor Radiator Area	18 m^2	13 m^2
Total Subsystem Mass	2145 kg	1341 kg
Total Required Power	97.7 kWe	98.3 kWe

There are many engineering choices and options that must be evaluated when a preliminary design of a total SP-100 ion engine system is undertaken.

4.3 SCIENCE INSTRUMENT RADIATION SENSITIVITY

The sensitivity tolerance of typical planetary science spacecraft instruments to SP-100 radiation was identified as a major concern in this study. In order to obtain a first order understanding of the issue, the instruments for the Galileo Project were selected for study. The Galileo instruments were selected for three reasons: (1) they are fully developed instruments about which a great deal is known, (2) the instruments have been specifically designed to accommodate the Galileo radiation environment (References 4-5 and 4-6), and (3) many of the Galileo instruments have been included in the payload for the Saturn Ring Rendezvous mission (see Subsection 3.3.2). The Galileo instruments are listed in Table 4-3.

Insofar as their radiation sensitivity is concerned, the Galileo instruments fall into two groups. Some instruments will be operating at or near their limits of radiation tolerance in the Galileo radiation environment. These instruments are generally the ones that involve light detection and include:

(1) Near Infrared Mapping Spectrometer (NIMS).

(2) Ultraviolet Spectrometer (UVS).

(3) Photopolarimeter Radiometer (PPR).

(4) Dust Detector (DDS).

(5) Solid State Imaging Science (SSI).

The other instruments are relatively immune to the effects of radiation and have no particular susceptibility other than that arising from their associated electronics packages. These instruments include:

(1) Magnetometer (MAG).

(2) Plasma Detector (PLS).

(3) Plasma Wave Detector (PWS).

(4) Radio Science (RS) experiment.

(5) Energetic Particle Detector (EPD).

Despite the fact that the Galileo Project is a major program and has involved a great deal of testing and documentation, there is remarkably little formal documentation available on the neutron and gamma ray tolerances of the payload instruments. Because of budget constraints, the Galileo Program operated on an announced philosophy of doing the minimum radiation testing necessary to assure mission success. The major radiation shielding challenge for the Galileo mission had to do with energetic electrons in the environs of Jupiter rather than with neutrons and gamma rays from the radioisotope thermoelectric generators (RTGs). Except for those few instances where neutron or gamma effects were a limiting factor, little formal testing was done and little formal documentation is available.

Most of the information in the following paragraphs was obtained from the individuals listed in Table 4-3. This subsection could not have been prepared without the contributions of these individuals.

Table 4-3. Galileo Instruments

Instrument	Cognizant Personnel	Institution
Solid State Imaging Science (SSI)	James R. Janesick Kenneth P. Klassen	JPL JPL
Near Infrared Mapping Spectrometer (NIMS)	Gary C. Baily Larry S. Varnall	JPL JPL
Ultraviolet Spectrometer (UVS)	Charles Hord	Laboratory for Atmospheric & Space Physics (LASP)
Photopolarimeter Radiometer (PPR)	Larry Travis Edgar Russel	Goddard Institute for Space Studies (GISS) Santa Barbara Research Center
Energetic Particle Detector (EPD)	Don Williams	John Hopkins University Applied Physics Laboratory
Dust Detector (DDS)	Martha S. Hanner C. J. Coppock	JPL JPL
Magnetometer (MAG)	Douglas Clay	JPL
Plasma (PLS)	Robert S. Wolff	JPL
Plasma Wave (PWS)	Robert S. Wolff	JPL
Radio Science (RS)	Dan F. Finnerty Joseph P. Brenkle	JPL JPL

Based on this survey of available documentation and conversations with cognizant persons, estimates of the radiation tolerance limits for neutrons and gammas for each of the Galileo instruments have been made. These estimates are presented in Table 4-4. The tabulated estimates are for instrument sensor elements themselves and not for any associated electronics.

4.3.1 Neutron Fluence Effects

Neutron effects testing for the Galileo instruments has been reviewed. Eight of the ten orbiter instruments were not tested at all for

Table 4-4. Estimated Neutron and Gamma Ray Tolerance for Galileo Orbiter Instruments

Instrument	Neutron[a] Fluence (N/cm2)	Gamma Ray Fluence (kRad)	Flux (Rad/sec)
Solid State Imaging Science (SSI)	$1 \times 10^{10} - 10^{11}$[b]	>10	0.001
Near Infrared Mapping Spectrometer (NIMS)	$\approx 10^{12}$[b]	20	0.09
Ultraviolet Spectrometer (UVS)	$>10^{13}$	10	0.02
Photopolarimeter Radiometer (PPR)	$>10^{13}$	30	0.3
Energetic Particle Detector (EPD)	$>10^{13}$	>500	0.000002
Dust Detector (DDS)	$>10^{13}$	≈120	≈1.2
Magnetometer (MAG)	$>10^{13}$	>500	≥10
Plasma (PLS)	$>10^{13}$	>500	≥10
Plasma Wave (PWS)	$>10^{13}$	>500	≥10
Radio Science (RS)	$>10^{13}$[b]	>500	≥10

[a]Flux probably $\geq 5 \times 10^4$ N/cm^2/sec, requires verification.

[b]Needs additional testing.

neutron effects because they were not believed to have any significant vulnerability to neutrons at the fluence levels anticipated for the Galileo mission (5×10^{10} N/cm^2). Persons familiar with seven of these eight instruments have affirmed that the instrument sensors could also be expected to function well at the much higher fluence levels anticipated for the SP-100 mission (as high as 10^{13} N/cm^2). The following three instruments are believed to be sensitive to the SP-100 plane radiation dose levels.

4.3.1.1 NIMS. Test results in the published literature (Reference 4-7) indicated that the InSb detectors used in the NIMS instrument might suffer significant signal-to-noise ratio degradation at neutron fluences of about 10^{12} N/cm^2. To assure proper operation for Galileo, the NIMS focal plane assembly was tested at a fluence of 3 x 10^{10} N/cm^2 (close to the anticipated Galileo exposure). There was no observable degradation. Further testing will be necessary if NIMS is to be used with SP-100.

4.3.1.2 SSI. The charge-coupled device (CCD) detector element for the SSI instrument was tested with neutron fluences up to 5 x 10^{10} N/cm^2. When the detector was operated at a temperature of -70°C, the instrument had an unacceptably high background signal level due to the creation of mid-band states. When the detector operating temperature was lowered to -85°C, the background noise was reduced to tolerable levels. On the Galileo flight, the operating temperature will be -110°C. Additional testing will be necessary in order to determine just how much additional neutron tolerance can be achieved through additional cooling. The 10^{10} - 10^{11} N/cm^2 figure in Table 4-4 reflects a disagreement among experts regarding the tolerance of certain noise levels and the efficacy of additional cooling.

4.3.1.3 RS. In the Radio Science (RS) experiment, it is considered possible, but unlikely, that neutron fluences of 10^{13} N/cm^2 could do enough displacement damage to cause significant shifts in the frequency of the ultrastable oscillator. Further testing would be required to clarify this matter.

4.3.2 Neutrons/Flux Effects

There are several mechanisms by which incident neutrons can produce spurious instrument signals at a rate proportional to the neutron flux. These mechanisms include (n, p) reactions, (n, α) reactions, and (n, γ) reactions. It appears that no rate dependent testing for neutron effects has been done on any of the Galileo orbiter instruments, and none of the cognizant personnel consulted regarding these instruments has indicated a need for such rate tests even at the higher flux levels anticipated for the SP-100 mission.

One may get an approximate numerical estimate of the importance of neutron flux effects by considering a somewhat typical silicon detector having an active volume one-centimeter square and 1/20-mm thick. For silicon, the three types of reactions listed above have an aggregate cross section of 0.011 barns. For the SP-100 mission, the neutron fluence would be accumulated at an approximately uniform rate during the seven-year period the reactor operates at full power. Thus, the maximum anticipated neutron fluence of 10^{13} neutrons/cm^2 corresponds to a flux of 4.5 x 10^4 (neutrons/sec)/cm^2. A straightforward calculation using these assumed values gives a maximum background corresponding to rate-dependent neutron effects. This computed background is 0.12 counts/sec, a result which suggests that the instrument experts are probably correct in saying that the rate-dependent effects are negligible for most or all of the Galileo orbiter instruments. On the other hand, this computed background is close enough to being significant to suggest that some testing is in order.

4.3.3 Gamma Ray Effects

The situation with gamma rays is both simpler and more complex than the situation with neutrons. It is simpler because the 500-kRad gamma ray dose anticipated for the SP-100 mission is less than one order of magnitude increase over the total radiation dose for Galileo (75 kRad). The neutron fluence increase was nearly three orders of magnitude. The situation with gamma rays is more complex because most of the 75 kRad for Galileo comes from energetic electrons. Gamma rays from the RTG represent only about 0.3% of the total dose.

Some instrument gamma ray tolerances cited in this subsection are inferred from tests done with energetic electrons as part of the Galileo program. This sort of inference from energetic electron data is possible because, to some extent, energetic electrons and gamma rays may be thought of as equivalent and interchangeable forms of radiation. When energetic electrons decelerate in matter, they produce gamma rays through a Bremmstrahlung process. Likewise, when gamma rays interact with matter, they can produce energetic electrons through processes such as Compton scattering.

Gamma rays effects, like neutron effects, fall into two classes: fluence effects (corresponding to permanent damage done by the gamma rays) and flux or rate-dependent effects (corresponding to the arrival of individual gamma rays).

4.3.3.1 MAG, PLS, PWS, and RS. Four of the ten Galileo orbiter instruments are thought to be essentially invulnerable to gamma ray effects, even at the 500-kRad level for SP-100. These instruments are the magnetometer, the plasma and plasma wave instruments, and the radio science experiment. The detector element for the magnetometer is a simple flux coil and the detector for the plasma experiments is a simple dipole antenna. Neither of these detectors is vulnerable to gamma rays except at the most extreme intensity levels. Likewise, the radio science experiment is concerned with the propagation of radio waves through space, a process that is not affected by the presence of gamma rays.

4.3.3.2 EPD. The energetic particle detector (EPD) utilizes surface barrier detectors that are only about 5-microns thick. Individuals familiar with these detectors say that they do not suffer significant permanent damage (fluence) effects from gammas even at the 500-kRad dose level. On the other hand, the transitory (flux) effects from gammas are severe. The EPD was tested using the RTG as a gamma source (at the same distance that would apply in the Galileo flight), and it was found that the background count rate was just barely tolerable for the sorts of measurements that needed to be made in interplanetary space. This barely tolerable background corresponds to a dose rate of about 2×10^{-6} Rad/sec. For measurements made at a higher signal level (e.g., measurements made in the vicinity of a planet), a higher gamma ray background might be accommodated.

4.3.3.3 SSI. The virtual phase CCDs used in the SSI experiment were successfully tested with a gamma ray dose of 10 kRad as part of the Galileo Project and were successfully tested with a gamma dose of 1000 kRads in testing done outside the project. Rate effects were tested using energetic electrons, and backgrounds were found to be tolerable up to the point at which each pixel saw an average of five electrons during the exposure interval. In flight operation, these exposure intervals can be as long as sixty seconds. Acquiring this degree of exposure in this time interval can be shown to correspond to a radiation dose rate of about 0.001 Rad/sec.

4.3.3.4 NIMS, PPR, and UVS. The InSb and silicon detectors in the NIMS instrument have been tested with gammas and have been found to give good performance at doses up to 20 kRad and at dose rates up to 0.09 Rad/sec.

The Photopolarimeter Radiometer (PPR) has a measured gamma dose limit of about 30 kRad and a measured dose rate limit of about 0.3 Rad/sec.

The photoelectric detectors in the UVS instrument have a gamma dose limit of about 10 kRad. The limiting dose rate is found to be one that produces about 1000 counts/sec. This limiting dose rate can be shown to be equivalent to 0.02 Rad/sec.

4.3.3.5 DDS. The detectors used for the DDS are a channel electron multiplier and a charge sensitive amplifier. Both detectors are provided with some shielding for the Galileo application. Table 4-4 of this subsection presents an estimated gamma dose limit (1.2 Rad/sec). These estimates are based on fragmentary documentation associated with shielding design calculations. Although approximate, these figures suffice to show that the DDS is not one of the more vulnerable instruments so far as gamma radiation effects are concerned.

4.3.4 Conclusions

This study was concerned with the radiation tolerance of the sensor elements for each of the ten Galileo orbiter instruments. The particular point of concern was whether these sensor elements could function effectively in the neutron and gamma radiation environments likely to be encountered on an SP-100 planetary mission. Attention has been paid to both the permanent damage effects associated with total accumulated dose (fluence) and to the transient, rate-dependent, spurious background count effects associated with dose rate (flux). The general conclusion reached is that most of the Galileo orbiter instrument sensor elements can be expected to function quite well in the neutron and gamma ray environments anticipated for the SP-100 planetary mission.

Currently available data and expert opinion suggests that eight of the ten orbiter instruments have neutron fluence tolerances that are in excess of the 10^{13} neutrons/cm^2 level presently regarded as worst case for the SP-100 mission. The neutron fluence tolerances for the SSI and NIMS instruments are estimated to be 10^{11} and 10^{12} neutrons/cm^2, respectively. These estimates are order of magnitude figures. Further testing is needed.

No rate-dependent neutron effects testing was done for Galileo because these effects were thought to be negligible at Galileo flux levels. Computations based on reaction cross sections suggest that neutron rate effects will probably (but not definitely) still be negligible at the higher flux anticipated for SP-100.

All the neutron tolerance estimates are based on limited data and rather lengthy extrapolations. An extensive program of neutron effects testing will have to be undertaken if Galileo instruments are to be flown with confidence on an SP-100 mission.

Some Galileo instrument sensors have been tested directly with gamma rays. In other cases, gamma tolerance limits have been inferred from tests done with energetic electrons.

Depending on the spacecraft and mission configuration selected, the SP-100 mission may involve instrument gamma ray exposures ranging from 25 to 500 kRad. Six of the ten Galileo instruments have gamma fluence tolerances in excess of 500 kRad. None of the other four instruments have tolerances that are much under 25 kRad. The most vulnerable of the instruments, the UVS, has an estimated tolerance of 10 kRad. With modest shielding the UVS could operate in the 25-kRad environment.

Eight of the ten orbiter instruments have gamma flux tolerances that are significantly higher than the flux anticipated for the SP-100 mission. The SSI has a flux tolerance close to that anticipated for the mission. The final instrument, the EPD, would probably not be able to function effectively in the low signal regime associated with making measurements in interplanetary space, but it might still give meaningful measurements in the higher signal regime associated with a planetary encounter.

4.4 ATTITUDE AND ARTICULATION CONTROL

Due to two nuclear electric propulsion (NEP) spacecraft design constraints, i.e., (1) low payload radiation tolerances and (2) strong reactor radiation, it was recognized that the spacecraft would be long (to reduce radiation dose at the payload without very massive shields). The resultant trial designs illustrated that attitude and articulation control would be a major spacecraft design challenge. The following paragraphs describe the steps by which the attitude and articulation control system (AACS) was developed and its final form.

4.4.1 NEP ACCS Functional Requirements

The functional requirements are about the same for any interplanetary spacecraft. But for NEP, the emphasis is on control and stability of a highly flexible vehicle being accelerated for long periods by low thrust electric propulsion. The following list of AACS functional requirements focus on the special AACS needs of electric propulsion.

- Point science to the required knowledge and stability.
- Provide a celestially based thruster vector pointing.
- Measure spacecraft acceleration magnitude and direction.
- Point the thrust vector through the spacecraft center of mass (CM).
- Provide three-axes control during nonthrusting periods.
- Point high gain antenna (HGA) to Earth.
- Provide near target autonomous navigation.
- Provide multipayload accommodation with minimum design changes.

4.4.2 Operation and Implementation Design Constraints

Functional requirements typically may be achieved with many designs, most of which are unacceptable for some mission peculiar operational or configuration constraints. The following operational goals and configuration constraints are a means of assisting spacecraft design to provide both functional and operational mission compatibility with the least number of design iterations.

- Minimize propulsion contamination of the spacecraft, instruments, and sensors.
- Use gyro stabilized thrusting.
- Use a dedicated three-axes inertial reference unit (gyros and accelerometers) rigidly attached to the thruster module.
- Require zero center of mass motion due to thrust vector direction control.
- Use the mercury tank as a spacecraft radiation shield.
- Require minimum boom deformation sensing.
- Require no propellant slosh.
- Require no active boom deformation control.
- Minimize attitude control effects of a large number of failed thrusters.
- Require no spacecraft turns for nonscience such as optical navigation.

- The science pointing will provide smear free images in the environment of ion thruster acceleration or large spacecraft flexible body motion.
- Maximize propulsion efficiency by minimizing mutual thruster force cancellation.
- Minimize thruster control gimbal angles.
- Ion thrusters to provide three-axes control torque during thrusting.

4.4.3 Configuration Options

Three spacecraft configurations are proposed here. Each configuration satisfies many of the configuration design constraints; however, none satisfy all. Each configuration description will list the major design assets and liabilities of each option.

4.4.3.1 Option a. This configuration is illustrated in Figure 4-1(a). The SP-100 is at one end of the boom and the spacecraft at the other end. In the middle is the mercury tank and thruster module.

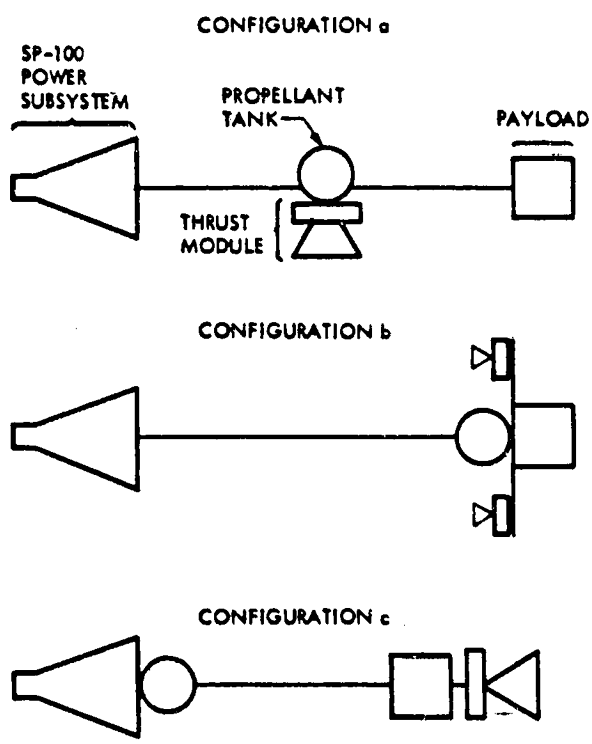

Figure 4-1. NEP Spacecraft Configuration Options

4.4.3.1.1 Assets.

- Mercury consumption does not markedly effect the vehicle center of mass relative to the thrust vector.
- Short high-power cables.
- Large thruster to payload separation distance.

4.4.3.1.2 Liabilities.

- Small thruster control torque about the largest two inertia axes.
- Gyros and accelerometers required at thruster module.

4.4.3.2 Option b. This configuration is illustrated in Figure 4.1(b). The SP-100 is at one end of the boom and the mercury tank, spacecraft, and thruster assembly at the other.

4.4.3.2.1 Assets.

- The vehicle is passively stable with the center of mass being pulled by the thrusters.
- Control torque about the thrust axis is relatively large.

4.4.3.2.2 Liabilities.

- The SP-100 subsystem is in the path of the thruster exhaust beam.
- Complex thruster deployment.
- Vehicle stability vulnerable to thruster module deployment or thruster failure.
- Long high-power cables.

4.4.3.3 Option c. This configuration (Figure 4-1(c)) has the SP-100 at one end of the boom, the spacecraft and a single thruster module at the other. The mercury tank may be placed at any position on the boom length, but from an AACS point of view, the nearer the SP-100 the better.

4.4.3.3.1 Assets.

- Configuration stability least vulnerable to thruster failure.

- Large control torques about vehicle's large inertia axes with the smallest thrust vector direction change.
- Most efficient thrusting.

4.4.3.3.2 Liabilities.

- Low thruster control torque about the thrust axis.
- Infrared reactor position sensor required on the spacecraft for thrusting stability.
- Long high-power cables.
- Short separation distance betwen thrusters and payload.

All the proposed configurations violated one or more of the requirements or constraints. The spacecraft configuration with the thruster axis parallel to the boom axis was considered more easily controllable than other systems. However, Configuration a was selected on the basis of the desire to keep the payload as far away as possible from both the reactor radiation and the ion thrusters' plasma environment.

4.4.4 Center of Mass Thrusting Configuration

The following analysis was used to size the center of mass thrusting configuration parameters. It was assumed that all three-axes control torques are provided by thrusters during thrusters use. The largest spacecraft disturbance torque occurs after spacecraft boom deployment in a 700-km orbit. This torque is the gravity gradient torque in the worst case orientation of the boom axis at 45 degrees to the local vertical axis. (Even if normal operations did not include this orientation, recovery from this worst case orientation is a requirement.) The worst case torque is evaluated in Table 4-5 for both Earth and Saturn as follows:

$$\text{Torque}_i = 3K(I_j - I_k)/2R^3 = \text{Torque about i-axis}$$

I_j = Inertia about the j-axis

R = Planet center to spacecraft mass center

K = Constant

4.4.4.1 Center of Mass Thrusting Configuration Design.

Design Requirement:

(1) Control torques must exceed disturbance torques.

Table 4-5. Worst Case Torque

Parameter	Earth	Saturn	Unit
K	4.92×19^{14}	3.68×19^{16}	m^3/s^2
R	6378 + 700	60400 + 6040	km
Large inertia axis torque	9.2	1.1	N-m
Small inertia axis torque	1.0	0.1	N-m

Design Goals:

(1) Thrusters must be used in pairs and produce both torque polarities in two axes and one polarity in the third.

(2) Thruster throttling for attitude control should be minimized.

(3) The spacecraft mass center must be moveable along the axis orthogonal to the thrust axis.

(4) The number of thruster gimbals is to be minimized and all gimbal travel limited to ±30 degrees.

Figure 4-2 conceptually illustrates a spacecraft thruster configuration that closely approaches all design goals. There are two pods of tightly packed thrusters arranged parallel to the boom axis one meter from the center of mass axis along the antithrust axis. The thrusters on the boom axis are about seven meters from the center of mass.

4.4.4.2 Operation. Control torque about the X-axis is achieved by engine thrust throttling or center of mass movement along the Y-axis.

Control torque about the Z-axis is achieved by differentially gimballing the entire thruster groups about an axis parallel to the Y-axis.

Control torque about the Y-axis is achieved by equally gimballing the entire thruster groups about an axis parallel to Y-axis.

4.4.4.3 Salient Design Features. The unit consisting of thrusters, propellant tank, power processing units, boom canisters, etc., is considered rigid. The control torques are not produced by couples. The spacecraft translations produced by noncoupled torques are easily compensated by the

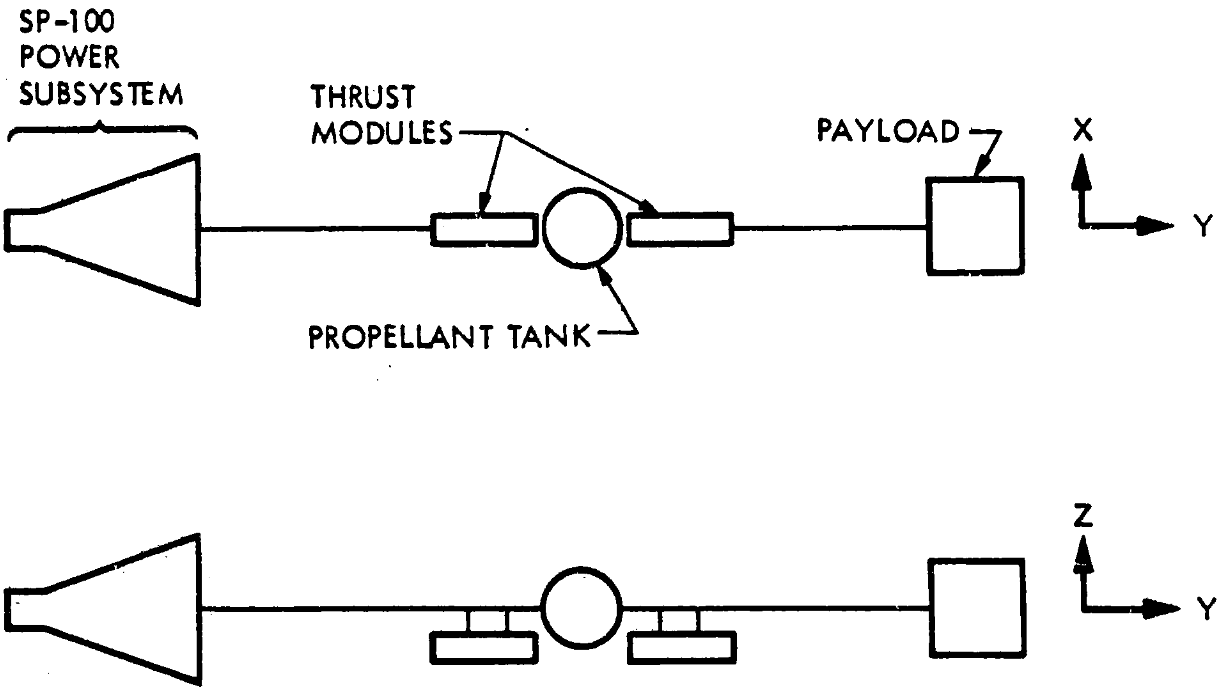

Figure 4-2. Side Thrusting, Center of NEP Spacecraft Configuration (the thrust vector is parallel to the Z-axis and in the positive direction)

continuous thrusting of the ion thrusters. The number of gimbals for full three-axes control is <u>one per thruster pod</u>. Control torques are decoupled in all axes.

4.4.5 Chemical Reaction Control System

A reaction control system (RCS) using chemical propellants is needed to maintain control of the NEP vehicle during the following periods: (1) deployment (before SP-100 and ion engine startup), (2) periods of coast (ion engines turned off), and (3) as a backup and enhancement of the control authority provided by the ion engines during periods of unexpectedly large disturbance torques. The requirements on the RCS listed below are aimed at maximizing functional attitude control and reliability but minimizing structure and complexity.

(1) RCS attitude control torques are couples that minimizes delta-V corrections and ion engine startups.

(2) Reliability requires two redundant hydrazine RCSs, each capable of providing the required attitude control torques.

(3) Each thruster assembly will provide up to seven thrust levels in each control direction that will permit shaping control torque impulse rise and fall characteristics compatible with flexible vehicle structure and attitude control stability requirements.

(4) There will be a minimum of plume impingement on the spacecraft.

(5) There will be an active propellant conservation control policy that reduces RCS control torque until one sided limit cycle occurs, or the lowest thrust level is reached.

4.4.6 RCS Design

All of the above requirements are satisfied by two redundant RCS systems, including two separate tanks, which are part of the nonrotating ion engine support structure. The thruster modules on each tank are mounted on two single-hinge booms, which are part of the tank support structure. Each thruster module should have at least three separately commandable thrusters in each of five thrust directions with nominal relative magnitudes 1, 2, and 4. The thrust direction of each thruster is parallel to one of the three orthogonal control axes.

Three thrusters (with different fixed thrust levels) can provide seven different thrust levels when used in combinations. A large selection of fixed thrust levels is a low cost method of programming control torque impulse rise and fall characteristics. Being able to select, for a range of circumstances, the rise and fall characteristics of the control torque impulses allows the control system to be most compatible with a large, flexible spacecraft such as the NEP Saturn Ring Rendezvous spacecraft. A wide range of magnitude of control forces and torques is desirable since large magnitudes are required to overcome large disturbance torques and low magnitudes are required in order to conserve propellant during long, low activity coast periods.

4.4.7 Articulation System (Payload Pointing and Attitude Determination)

The previous subsections have described the baseline configuration, i.e., SP-100 at one end of a long boom structure, center of mass thrusting ion engines at the spacecraft center of mass, and the payload at the other end. The long boom distance between the SP-100 and the payload is required for radiation protection, but a long boom is an extremely flexible member. Disturbances at the reactor end of the boom, such as Stirling engine disturbances, will be observed at the payload end of the boom, but the magnitude and the frequency relationships between reactor disturbance and payload response have not been determined yet. However, it may be stated with certainty that if the payload adds energy to the boom deformation by applying forces and torques to the boom in phase with the deformations, there will be boom and stability problems. This energy reinforcement typically results when a gyro stabilized, Galileo type science platform removes base vehicle motion in pointing the platform.

4.4.8 NEP Payload Pointing Requirements

The payload pointing system will point the science instruments and the celestial sensors for the payload and spacecraft attitude determination.

Because of the unique NEP missions, spacecraft configuration, and operational modes, many sensor systems may be required to monitor the status of spacecraft structure, thrusters, reactor, etc. It is assumed here that to minimize special sensor development, the science instruments, especially the TV cameras, will be used for the necessary observations. Platform instrument radiation protection is provided by shields on the platform and operational limits on SP-100 observation time.

The platform pointing requirements below reflect science and spacecraft observational needs.

(1) Point science to the required accuracy and stability.

(2) Provide pointing capability that is compatible with large flexible spacecraft.

(3) Provide a large platform viewing field including the SP-100, thrusters, and boom.

(4) Provide platform celestial sensors adequate for both science and spacecraft attitude determination.

(5) Provide on-board navigation compatibility.

(6) Provide new technology compatibility with little design impact.

4.4.9 Pointing System Design

The platform pointing concept required is a two degree of freedom, gyro stabilized, momentum compensated platform, with the rotation axes passing through the platform mass center, and all spacecraft and science attitude determination sensors mounted on the platform. The concept is called the integrated platform pointing and attitude control subsystem (IPPACS). The IPPACS concept and performance are summarized below. IPPACS is the baseline AACS for the Mariner Mark II spacecraft. IPPACS may be mounted on a long boom at any place on the spacecraft, subject only to radiation and boom linear acceleration constraints.

4.4.9.1 IPPACS Description. The IPPACS is part of a two degree of freedom, momentum compensated, inertially stabilized platform, with both axes of revolution through the platform mass center. Also included on the platform are a target body star tracker (TBST), the equivalent of an ATAC-16 processor, and all pointed science instruments as illustrated in Figure 4-3.

The attitude control function is reduced to determining the celestial orientation of the platform (the TBST gives three axes of platform angular position from one multiple star pattern) and the IPPACS processor commands the spacecraft torquers to move the spacecraft relative to the IPPACS platform to achieve the desired spacecraft-Earth/Sun pointing.

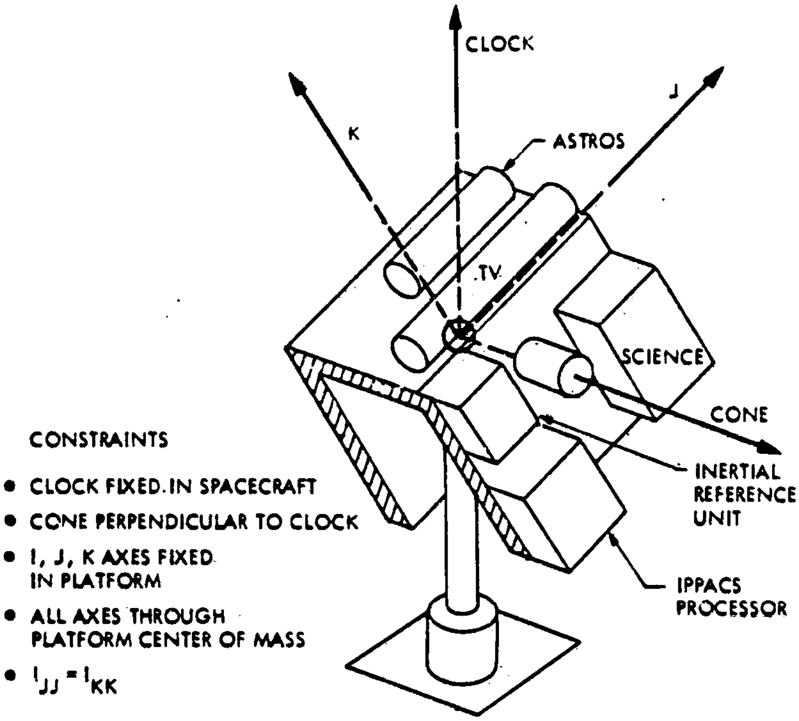

Figure 4-3. The Integrated Platform Pointing and Attitude Control Subsystem (IPPACS) Configuration

The IPPACS spacecraft operational modes include:

(1) Celestial references (TBST) only used in conjunction with analytic damping (rate estimation) during long cruise periods.

(2) Inertial (gyros) only using inertial rate plus positions updated periodically to eliminate drift with celestial TBST data. This mode is used during high resolution pointing.

(3) Inertial (gyro) rate plus celestial (TBST) position. In any orientation, this mode is rarely achievable on current technology spacecraft because the desired inertial reference direction rarely has celestial objects in the spacecraft fixed celestial sensors fields' of view.

The implications of a momentum compensated platform are very important for pointing performance and cost, and are discussed as follows:

(1) The IPPACS dynamically isolates the platform from the spacecraft, making spacecraft and mission design independent of instrument pointing and attitude control design.

(2) Since the IPPACS design can achieve all projected pointing requirements, it is independent of pointing requirements, and can be standardized for all missions requiring platforms.

In summary, a few of the IPPACS key features are listed below:

(1) The key elements of the ACS and platform pointing subsystems are located on a single structure at the focus of the most rigorous mission pointing tasks. All pointed science instruments are also part of this structure.

(2) Arc second pointing stability requires an inertial reference set (gyros) on the pointed platform to achieve the desired disturbance sensing bandwidth and damping.

(3) The most versatile position for the TBST for both closed loop star tracking, instrument calibration, autonomous or ground supported optical navigation, or target signature tracking is on the science platform.

(4) High data rates between the TBST, gyros, actuators, and processor require that they all be on the platform.

(5) Since the platform is mass center mounted and momentum compensated, it can remove disturbances originating on the spacecraft without introducing disturbances into the spacecraft, therefore permitting more flexibility in spacecraft design than would normally be permitted. In addition, the momentum compensation feature permits generalized pointing algorithms to be developed and and eliminates or greatly reduces one of the major mission costs: nonrecurring hardware design and pointing analysis and control software development.

(6) The platform position sensor (encoder) resolution and cost are now driven by antenna pointing and other low accuracy attitude and control (A/C, functions.

(7) The IPPACS is fully testable on Earth.

4.4.9.2 IPPACS Performance. The IPPACS performance of the current Mariner Mark II design has an accuracy of 0.001 radian and a stability of 2 arc seconds. Table 4-6 presents the AACS subsystem equipment list.

4.5 TELECOMMUNICATIONS

The assumptions for the telecommunications link design are as follows:

(1) Data Rate: 268.8 Kbps (same as the Venus Radar Mapper mission).

(2) Range: Up to 10.5 AU.

(3) Frequency: X-band, 8415 MHz.

Table 4-6. AACS Equipment List

Description	Number of Units	Location of Unit*	Mass Total kg
Processor	2	IP	3.0
Memory	2	IP	4.0
I/O Electronics	2	IP	8.0
Power Supply	2	IP	2.4
Sensors			
Star and Target Body Tracker	2	IP	18.0
Sun Sensor	2	PM	4.0
Accelerometer (Mesa)	4	TM	8
Accelerometer Electronics	2	TM	4
IRU (Mech. 3 Axis)	1	TM	10
IRU (Fors 3 Axis)	1	IP	10
Actuators			
Platform Actuator	2	IP	14
Actuator Electronics	2	IP	10
Flywheel	2	IP	15
HGA Actuator and Driver	2	PM	2
Ion Engine Gimbal Drivers	2	TM	2
Ion Engine Throttle Driver	2	TM	2
Boom Actuators and Drivers	2	TM	2
RCS Thruster Drivers	2	TM	2
Spinning Science Actuator and Driver	1	PM	2
Total			122.4

*TM = Thrust module. PM = Payload module. IP = IPPACS platform.

(4) Ground Stations: DSN 34-m high efficiency (HEF) or 70-m at Canberra.

(5) Ground Elevation Angle: 25 degrees.

(6) System Noise Temperature: 34- and 70-m stations 29.5 K.

(7) Ground Antenna Gain: 34-m/67.5 dBi; 70-m/74 dBi.

(8) Bit Error Rate (BER): 5×10^{-3}.

(9) Spacecraft Antenna Pointing Loss: -1.00 dB (independent of HGA size).

The system design followed work done in this area for a previous study (References 4-8 and 4-9).

There are two options for the spacecraft power amplifiers. One is the use of large traveling wave tube amplifiers (TWTAs) - redundant units would be required. The other would be the use of an array feed power amplifier (AFPA) as described in Reference 4-9. The AFPA has built-in redundancy as it is composed of 88 elements, each element being a 5-W X-band solid-state power amplifier (XSSPA).

If TWTAs are used, they will be either mounted on the back of the high gain antenna (HGA) or be in a spacecraft bay. If they are in the bay, there will be significant circuit loss between the TWTAs and the HGA. The HGA is on a boom (5 m). It is assumed here that this loss is 3 dB. If the AFPA is used, the circuit loss will be much less, about 0.3 dB.

The DC to RF efficiency for the TWTAs is assumed to be 33%. For the AFPA, the DC/RF efficiency is assumed to be 30%.

Link calculations were performed (Table 4-7) for the following six options.

Option 3 was selected on the basis of being able to use the smaller DSN antenna and a small, solid 3.7-m high gain antenna on the spacecraft. Both of these items are conservative in that the smaller DSN antennas are more available and less costly than the 70-m antennas and a 3.7-m spacecraft antenna can be a copy of the antenna flown on the Voyagers 1 and 2 spacecraft.

Table 4-7. Option Calculations

Option	1*	2**	3	4	5	6
DSN Receiver, m	34	34	34	34	70	70
S/C Power Amplifier, W	100 TWTA	800 TWTA	440 AFPA	290 AFPA	200 TWTA	100 AFPA
High Gain Antenna Diameter, m	10.5	3.7	3.7	4.5	3.7	3.7

*Requires pointing error of 0.06 or less.

**It is questionable whether such a large TWTA (or parallel combination of TWTAs) would be developed.

Option 3 is enabled by use of the array feed power amplifier. The selected link design calculations are shown in Table 4-8. This design requires 1.47 kWe of DC power. The telecommunications equipment mass and power are summarized in Table 4-9.

Table 4-8. 34-m HEF/440-W AFPA/3.7-m HGA

	Design	Fav. Tol.	Adv. Tol	Mean	Variance
Transmitter Power, dBm	56.43	1.00	-1.00	56.4	.17
Circuit Loss, dB	-.30	.10	-.20	-.3	.01
S/C Antenna Gain, dBi	48.40	1.00	-1.00	48.4	.17
Pointing Error, dB	-1.00	.50	-.50	-1.0	.04
Space Loss, dB	-294.86	.00	.00	-294.9	.00
Polarization & Ground Pointing Loss, dB	-.20	.10	-.10	-.2	.00
Ground Antenna Gain, dBi	67.50	1.00	-1.00	67.5	.33
N_0, dBm/Hz	-183.90	.40	-.40	-183.9	.02
P_T, dBm				-124.1	.72
P_T/N_0, dB-Hz				59.8	.74
Carrier Threshold BW	20.00	-.04	.04	20.0	.00
Telemetry Suppression	-15.21	.82	-.90	-15.2	.12
Carrier Margin				24.5	.86
Data Rate	54.29	.00	.00	54.3	.00
Data Power/Total Power	-0.13	.01	-.01	-.1	.00
Systems Loss	-1.00	.30	-.30	-1.0	.02
E_b/N_0, req	2.30	.00	.00	2.3	.00
Margin				2.1	.76
Weather & Link Margin (90% Confidence)				2.1	

Table 4-9. Telecommunications Equipment Mass and Power Summary

Component	Inheritance	Mass Each, kg	Number	Total Mass, kg	Power, W
NASA X-Band Transponder	New	2.0	2	4.0	12
Command Detector Unit	New	0.5	2	1.0	---
X-Band Receiver Switch	VRM	0.25	1	0.25	---
Diplexer	VRM	1.0	1	1.0	---
X-Band Rotary Joint	New	0.3	2	0.6	---
X-Band Transfer Switch	VRM	0.25	1	0.25	---
X-Band Hybrid Filter	GLL/VRM	0.2	1	0.2	---
TMU	VGR	2.5	2	5.0	4.7
Interface & Control	VGR	2.0	1	2.0	1.0
X-Band 20-W SSPA	New	3.5	2	7.0	80.0
AFPA	New	74.0	1	74.0	1430
Relay Radio	GLL	9.0	1	9.0	---
Waveguide & Cabling	---	---	---	3.0	---
Total				107.3	
HGA	VGR	---	1	52.0	---
MGA	New	---	1	2.0	---
LGA	New	---	1	0.5	---
Probe Antenna	New	---	1	5.0	---
Total				59.5	

A block diagram of the telecommunications system is shown in Figure 4-4. A low gain antenna (LGA) is used for near Earth communications. The LGA is adequate for communications near the Earth and does not require precise pointing like the HGA. After leaving the vicinity of the Earth, the LGA will no longer be adequate and communications will be carried out by using the HGA with the X-band AFPA and a medium gain antenna (MGA).

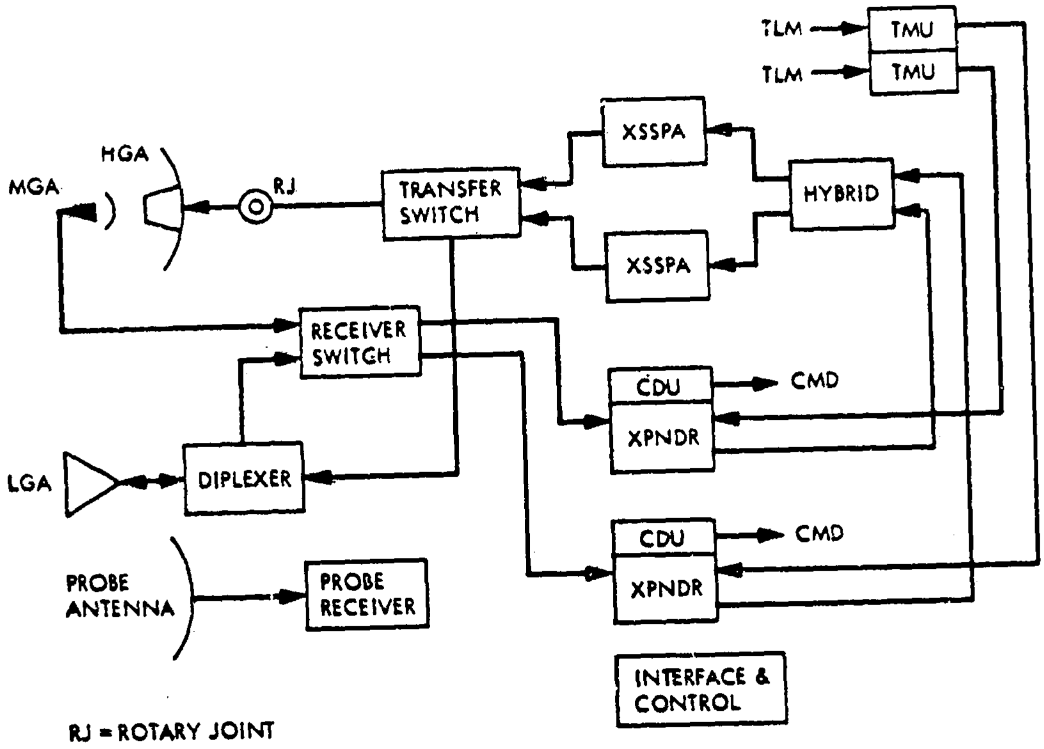

Figure 4-4. Telecommunications Subsystem Block Diagram

The HGA is for transmission only since the AFPA is not good for receiving. The MGA is required for reception of commands from Earth and for radio tracking. The MGA is attached to the HGA, which will provide more than adequate pointing for the MGA.

4.6 CHEMICAL PROPULSION

Two chemical propulsion issues were specifically dealt with during this study: (1) chemical stage delivery capability from shuttle orbit to nuclear safe orbit (NSO) and (2) the preliminary design of a chemical reaction control system (RCS) for the NEP spacecraft. As stated in Subsection 3.3.3, a small chemical propulsion stage is required to boost the NEP spacecraft from its shuttle parking orbit (278 km, 28.5°) to a NSO, which is assumed to be 700 km and 28.5°. Figure 4-5 presents the delivery capability of several small chemical propulsion systems. It is assumed that an entire shuttle payload, 29,500 kg, is the initial mass. This initial mass contains the small chemical stage plus its propellant, the payload to be delivered to NSO, and airborne support equipment (ASE), which remains in the shuttle payload bay. The chemical propulsion stages that were considered are: (1) Star 37G, 31, and 48 solid propellant motors (I_{sp} = 290 sec, burn-out mass = 100 kg), (2) a bipropellant, NTO/MMH, orbital maneuvering vehicle (OMV) (I_{sp} = 285 sec, burn-out mass = 1145 kg), and (3) a monopropellant, N_2H_4, satellite control system (SCS) (I_{sp} = 230 sec, burn-out mass = 1307 kg). The ASE is taken from Reference 3-2. The 3030 kg of ASE includes a 540-kg forward

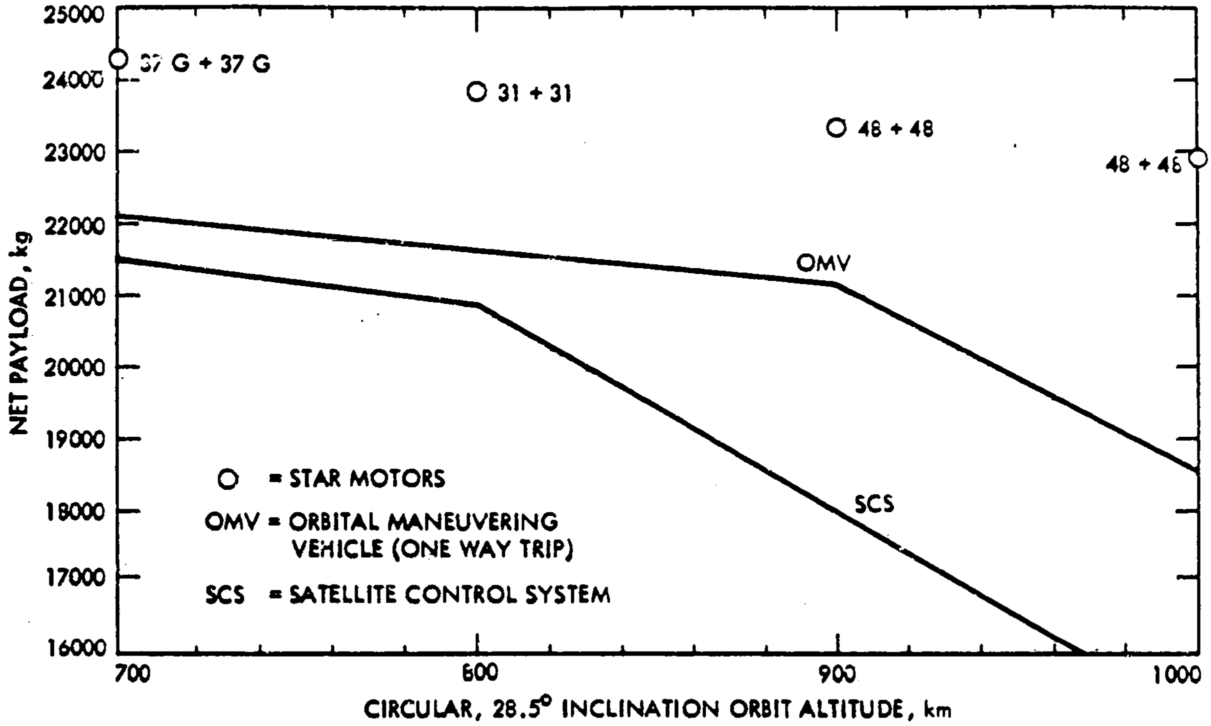

Figure 4-5. Delivery Capability of Several Chemical Propulsion from 278 km, 28.5°

cradle, a 1450-kg aft cradle, 605 kg for the remote manipulator system (for deployment), and 435 kg for miscellaneous and cradles for the chemical stage. The requirement is for roughly 17,000 kg to be placed into nuclear safe orbit (NSO). The OMV and the Star motors can deliver 22,000 and 24,280 kg, respectively, to 700 km and 18,500 and 22,960 kg, respectively, to 1000 km. Even if NSO is taken to be 1000 km, the Star motors and the OMV have excess of capability.

The attitude and articulation control Subsection 4.4 presented the requirements in the RCS and described a conceptual design. The following lines present a preliminary design that meets the requirements outlined in Subsection 4.4. The primary requirements on the RCS are that it provide three thrust levels of ratio approximately 1:2:4 in five directions at each RCS pod location. (The location of the four RCS pods on the propulsion module of the NEP spacecraft is shown later in Figure 4-8.) A hydrazine RCS was selected. The RCS thrusters are located on booms projecting from two spherical hydrazine (N_2H_4) tanks.

The RCS uses two spherical, titanium, positive expulsion propellant tanks for N_2H_4 storage. These tanks supply propellant over a blowdown ratio of 2:1 (i.e., initial pressure (P_i) = 350 psia, final pressure (P_f) = 175 psia). This tank pressure variation will result in a varying thruster inlet pressure (P_{in}). This forces a thrust level and specific impulse (I_{sp}) variation over the mission. However, the desired thrust ratio will be maintained because the thrust level variation will be similar for each thruster. The system will use existing thrusters at the following three thrust levels: 0.2, 0.5, and 1.1 lb_f. The ratio of these thrust levels is 1:2.5:5.5, and this system will provide the following seven thrust levels in each direction: 0.2, 0.5, 0.7, 1.1, 1.3, 1.6, and 1.8 lb_f. The specific impulse of these thrusters will be 160 sec for a short pulse and 210 sec for steady state. Table 4-10 presents the RCS mass summary. Two 22-inch diameter titanium tanks with positive expulsion diaphragm propellant management devices were chosen to contain the propellant while enabling a 2:1 blowdown ratio. A 10% mass contingency is included. With the assumed propellant loading of 50 kg per tank, the RCS wet mass is 243 kg.

4.7 CONFIGURATION AND MASS PROPERTIES

The configuration of the Saturn Ring Rendezvous spacecraft is shown in Figure 4-6. Figure 4-7 presents the spacecraft as it would appear from Earth in orbit around Saturn. Mass summaries for this spacecraft are presented in Tables 4-10 and 4-11, and mass properties are shown in Table 4-12. The configuration of the spacecraft is dominated by the requirement that the payload be positioned 65-m away from the reactor of the SP-100 power plant. This distance was selected to reduce the neutron and gamma radiation exposure to levels equivalent to the Galileo electronic component environmental requirement. The SP-100 power plant itself is divided into two sections: the reactor shield, power conversion, and the heat rejection radiator and, at roughly 25-m distance, the power conditioning electronics, shunt radiator, and system controls. The Saturn Ring Rendezvous mission has an additional large module; the electric propulsion module consisting of ion thrusters, mercury tank, power processing units (PPUs), and heat rejection radiators. The configuration of the SRR NEP spacecraft is then a pair of long booms, which position the components in line. The SP-100 reactor is at one end and the spacecraft at the opposite end, 65-m distant. The mercury tank is located 26.0 m from the reactor at the system center of gravity. The thrusters, SP-100 power conditioning equipment, and thruster power processing equipment are located at the mercury tank. The system is constrained to fit within the space shuttle cargo bay, a 4.6-m diameter, 18.3-m long cylinder.

4.7.1 SP-100

The SP-100 power plant is a nuclear reactor powered thermal-to-electricity conversion system rated at 100 kWe. The SP-100 system can be based upon static conversion methods such as in-core thermionic emission in the fuel elements or solid state thermoelectric elements or upon dynamic systems such as Brayton or Stirling cycles. The weights (masses) for the SP-100 listed in Table 4-11 are representative for all systems. The reactor generates the required thermal energy, which is removed from the core by an electromagnetically pumped liquid metal coolant to the energy conversion components. (In the case of the in-core thermionic emission, the conversion is in the reactor, and the liquid metal removes the waste energy directly to the radiator.) The waste heat from the energy conversion is then transported to heat pipe radiators for disposal to space. The reactor is positioned behind a gamma ray and neutron shield to reduce the radiation levels on the power conversion and heat rejection elements and the payload. The raw

Table 4-10. N_2H_4 RCS Mass Summary

Component	Number	Unit Mass, kg	Total Mass, kg
N_2H_4 Tank	2	8.17	16.34
Thruster Pods	4	3.00	12.00
Thruster (0.2 lb_f)	20	0.40	8.00
Thruster (0.5 lb_f)	20	0.41	8.20
Thruster (1.1 lb_f)	20	1.00	20.00
Pod Booms	4	6.5	26.00
Latch Valves	10	0.27	2.70
Pyro Valves	6	0.15	0.90
Service Valves	6	0.08	0.48
Filters	10	0.50	5.00
Tubing (length)	40 m	0.15/m	6.00
Brackets	100	0.05	5.00
Pressure Transducers	10	0.10	1.00
Temperature Transducers	10	0.01	0.10
Structure	2	5.00	10.00
Thermal Blankets	---	---	5.00
Residuals and Holdup	---	---	0.03 M_p
Subtotal			126.72 + 0.03 M_p
Contingency (10%)			12.67 + 0.003 M_p
Total (M_p = Propellant Mass)			139.39 + 0.033 M_p

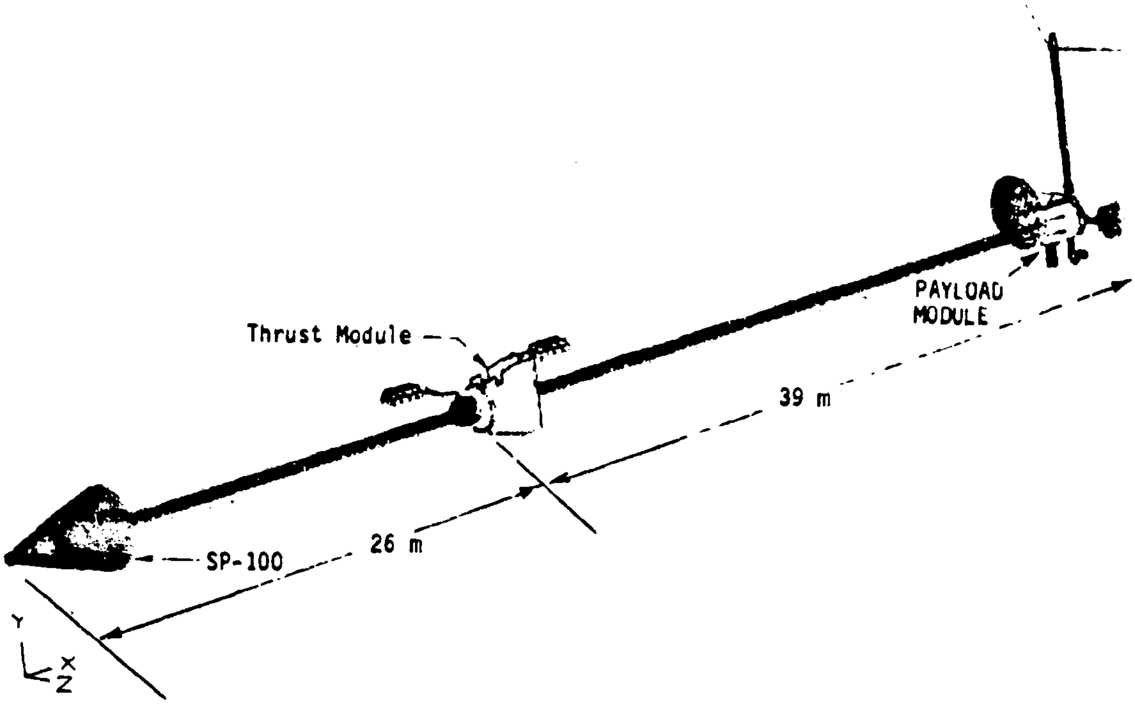

Figure 4-6. Nuclear Electric Propulsion Saturn Ring Rendezvous Spacecraft Configuration

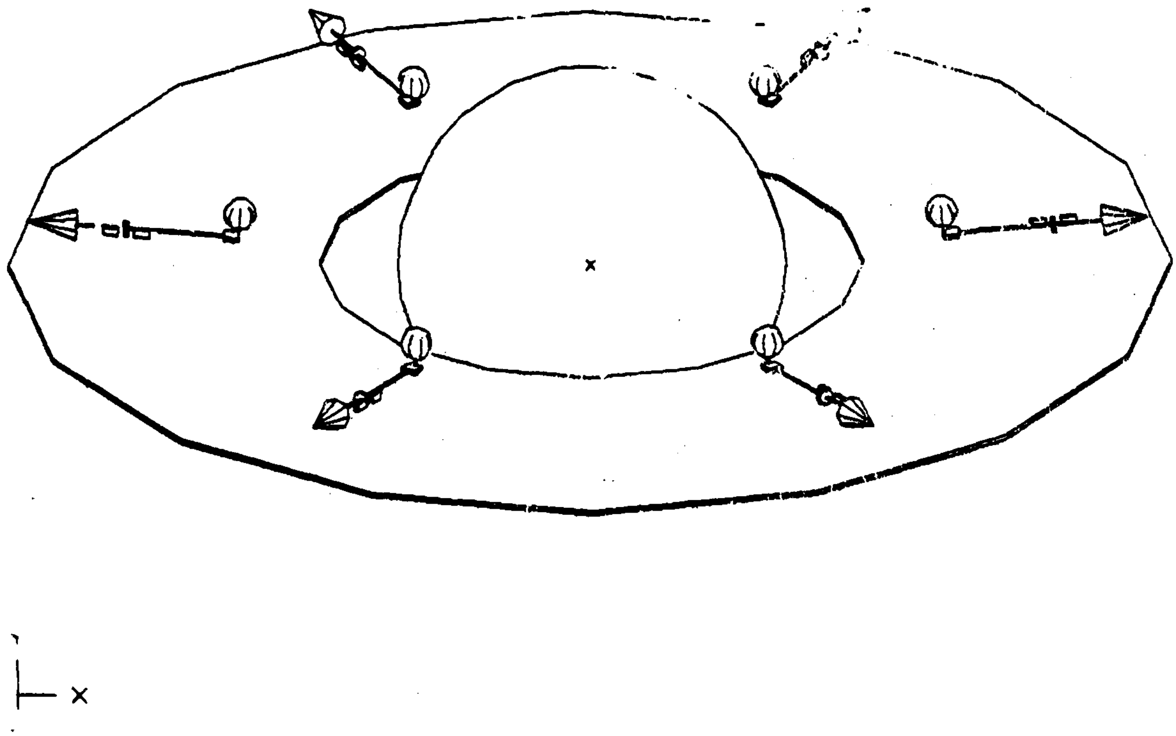

Figure 4-7. The Orientation of the Spacecraft Relative to Saturn as Viewed from Earth

Table 4-11. System Mass Summary (kg)

SP-100	Reactor	475
	Shield	660
	Heat Transfer	200
	Power Conversion	430
	Radiator	530
	Structure	150
	Power Conditioning	455
	Boom and Cable	100
	Subtotal	3,000
Thrust Module	Ion Thrusters	192
	Power Processor Units	592
	Thermal Control	376
	Structure and Actuators	241
	Power Distribution	524
	Harness and System Controller	56
	Electric Propulsion Contingency	64
	Mercury Tank	100
	Mercury	10,386
	Dry RCS	126
	Hydrazine	103
	RCS Contingency	13
	Attitude and Articulation Control	30
	Subtotal	12,803
Payload Module	Structure	737
	Mechanical Devices	39
	Thermal Control	38
	Harness	60
	Pyro	11
	Attitude and Articulation Control	92
	Telecommunications	107
	Antenna	60
	Command and Control	34
	Data Storage	18
	Power	100
	Science	
	Imaging	53
	Particles and Fields	38
	Remote Sensing	85
	Titan Probe	250
	Boom	120
	Subtotal	1,842
	Total Wet Spacecraft	17,645

Table 4-12. SRR Spacecraft (Deployed) Mass Properties (Referenced to Figure 4-6)

Surface Area = 412.6 m^2
Volume = 24.0 m^3
Mass = 17,645.0 kg

X, Y, Z of Center of Mass: 6.73 m, 0.016, 0.0096

Principal Moments of Inertia about Center of Mass

IAA = 1.53×10^4 kg·m^2
IBB = 4.20×10^6
ICC = 4.20×10^6

Moments of Inertia about Center of Mass Referenced to Entity Axes

IXX = 1.53×10^4 kg·m^2	IXY = -1.41×10^4
IYY = 4.20×10^6	IYZ = -4.72×10^1
IZZ = 4.20×10^6	IZX = 4.33×10^3

Rotation Matrix to Principal Axes from Entity Axes

AX,AY,AZ: 1.0000, -0.0034, 0.0010
BX,BY,BZ: -0.0034, -0.9999, 0.0097
CX,CY,CZ: -0.0010, 0.0097, 1.0000

Rotation Angles to Principal Axes from Entity Axes

Angle about X = 0.55°
Angle about Y = 359.94°
Angle about Z = 359.80°

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electrical power is conducted to the power conditioning equipment, located roughly 25 m from the reactor, for voltage regulation and matching the load demand. The system controls are also located at this remote location because of vulnerability of electronic components to the high radiation environment closer to the reactor. The boom between the two halves of the SP-100 is 24.0-m long and stows in a length of 1.4 m. The shunt radiator, used to dissipate excess electrical power at 1100 K, is located around the boom stowage cannister. A power cable is used to deliver the raw electrical power to the power conditioning equipment.

4.7.2 Chemical Reaction Control Assemblies

The reaction control tankage and reaction control thrusters (see Figure 4-8) are divided into two discrete systems that operate together to give attitude adjustments as couples; but either system can act alone, if

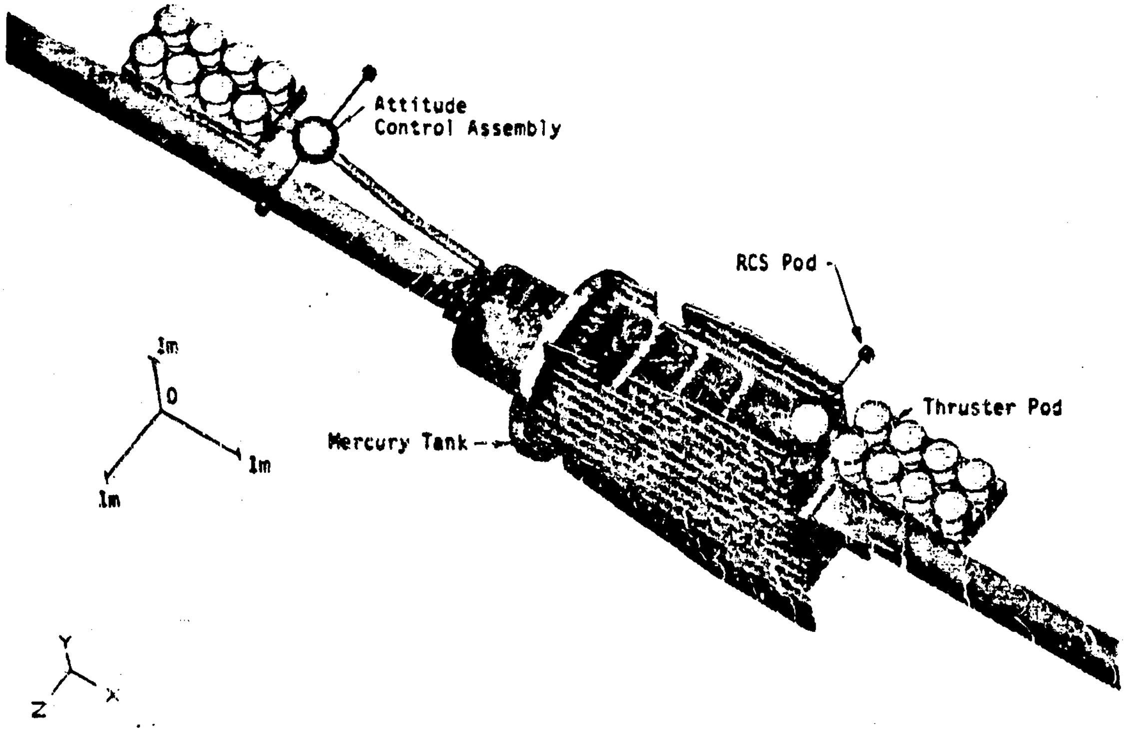

Figure 4-8. The Electric Propulsion and Attitude Control Module

required, offering system redundancy. In addition, the rear attitude control system is used to provide attitude control stabilization following separation from the shuttle and prior to deployment of the spacecraft to the cruise configuration. Each attitude control module consists of a central hydrazine tank and reaction control thruster pods on the ends of two opposed struts. The struts are rigidly attached to the tank and no deployment or flexible joints are required in the feed system. The thrusters are aligned with the ±Y, ±X, and the ±Z axes. The only spacecraft structure in the thruster plumes would be the power processing unit (PPU) radiators in the -Y plumes. The thrusters are each a grouping of three units with relative thrusts of about 1:2:4 to allow the level of thrust to be selected as required. This will also allow the thrust to be built up gradually to avoid structural resonances when the reaction control system is pulsed. The ion thruster array actuators are mounted on the reaction control system hydrazine propellant tanks.

The forward reaction control system and the forward ion thruster array fold back between the PPU radiators in the system stowed configuration. The booms for the reaction control thruster pods fit into cutouts in the PPU radiators. The power and mercury lines for the thrusters cross the deployment pivot on the forward array boom and the actuator articulation axes of both forward and rear thruster arrays.

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2 of 2

4.7.3 Electric Propulsion Module

The electric propulsion module (see Figure 4-8) is positioned so that the mercury tank is at the spacecraft center of gravity. The 10,386 kg (which includes 300 kg or 3% for residual and hold-up) of mercury represents 60% of the spacecraft mass, and placement of the tank at the system center of mass ensures that the CM will not migrate as the propellant is used. The mercury tank, with a volume of 0.7 m^3, is shaped as a disk 2.2-m in diameter and 0.3-m thick. It is assumed that the tank has internal features that keep the mercury spread in a uniform thickness across the tank diameter as the mercury is used. The tank shape and mercury management is intended to allow all components "down-stream" of the tank to take advantage of the additional gamma ray attenuation afforded by the mercury. The SP-100 power conditioning and controls electronics are located behind the mercury tank as are the eight power processor units and the electric propulsion controls. These components are all mounted between two radiator panels used for dissipation of waste heat. The radiator panels may require louvers for thermal control since the heat to be rejected may vary widely.

The 100-kW electrical power output of the SP-100 limits the operation of the 0.72-newton thrusters to only four of the 16 units at any one time. The thrusters are arranged in two arrays of 2 x 4 thrusters. The arrays are displaced 5-m fore and aft of the mercury tank and 1-m above the boom centerline. Each array is a structure supporting the eight thrusters. Each array is provided a single degree of freedom about the roll axis by an actuator attached to the reaction control tank. Attitude control of the spacecraft during thruster operation is intended to be solely by manipulation and throttling of the thrusters using supplemental reaction control system operation as required. Roll and yaw control is provided by pivoting the thruster arrays together or in opposition; and pitch control is provided by throttling thrusters.

The PPU radiator plates are aligned parallel to the ion thrust axis to minimize solar heating. The storage and deployment cage for the rear boom is located between the PPU radiator plate.

4.7.4 Payload

The payload (see Figure 4-9) includes everything to the rear of the electric propulsion module. The 38.2-m long boom stows in a 3.5-m container/ deployment mechanism between the PPU radiators. The boom deployment mechanism may have the capability for boom length adjustment during the mission to fine tune the center of mass location to enhance attitude control characteristics.

The science is mounted on a (2.0 x 1.8 x 0.5 m) bus at the far end of the boom. Power to the science and the bus is carried in a small cable on the boom. The science is separated into four areas: a magnetometer mounted on the bus, a Titan probe, instruments that sweep a conical path from a spinning mount, and instruments that are pointed at targets of interest from a two degree of freedom scan platform.

The bus communicates to Earth through a 3.66-m diameter rigid antenna of the Voyager HGA type. The antenna is stowed on this spacecraft

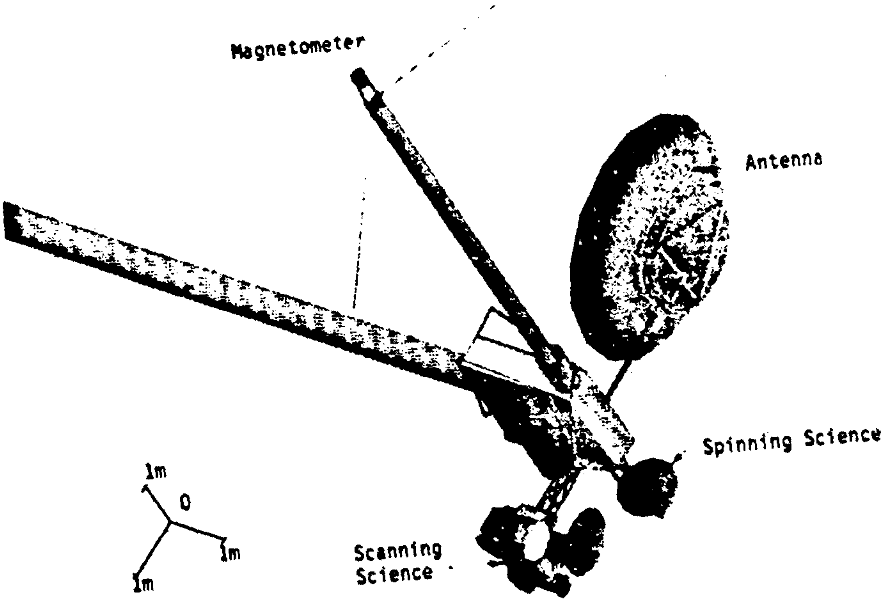

Figure 4-9. The Payload Module

with the radio frequency (RF) axis perpendicular to the spacecraft centerline. This results in the bus being only 0.50-m thick to allow the antenna to fit within the STS bay. Stowing the HGA on the spacecraft axis is precluded by the need to attach the propulsion system for transfer to 700 km on this end of the spacecraft.

The placement of the payload components relative to Saturn's ring during the ring rendezvous phase is shown in Figure 4-10. The scan platform is mounted on the wide face of the bus closest to Saturn's rings. This permits the science to view the rings and the surface of Saturn with the least obstruction. The scan platform boom must fold to reduce the science height to fit within the shuttle bay. The spinning science is mounted on the narrow bus face opposite to the ion thruster beams to minimize mercury contamination. This allows the spinning science to see the planet, the rings, and space. The magnetometer is mounted on the opposite narrow bus face to preclude spinning science blockage. The magnetometer must fold over against the bus to provide clearance for the ion thruster array when the spacecraft is stowed. The Titan probe is mounted adjacent to the spinning science.

The antenna is mounted on the wide side of the bus away from the rings. The antenna is deployed by two arms that form a pylon upon unfolding. The antenna mounts on a triangular frame that interfaces with a two degrees of freedom actuator on the end of the pylon. The antenna will have to continuously move in both axes to track Earth since the spacecraft is Saturn oriented.

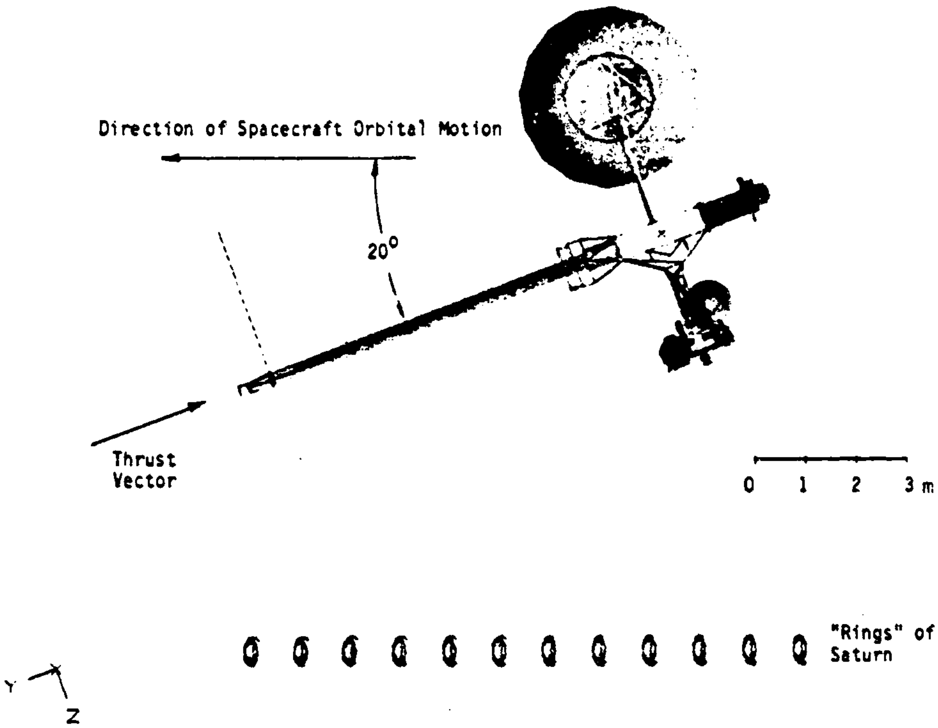

Figure 4-10. The Orientation of the Spacecraft Relative to Saturn's Rings as Viewed Along a Radius Connecting the Spacecraft to the Center of Saturn

Saturn has an orbital period of 29 1/2-years. The rotational axis of Saturn is inclined 26° 7' to the ecliptic plane. The orbit is eccentric so that the period during which the north pole is toward the Sun is 15 3/4-years while the south pole is toward the Sun 13 3/4-years. The rings will have maximum exposure (north face illuminated) in 2001, edge on to the Sun in 2009, and maximum (south face illuminated) in 2017. It is assumed that the trajectory will place the spacecraft on the sunlit side of the rings. This prevents the antenna from having to see Earth through the rings and reduces the scan range in negative elevation.

4.7.5 Stowed Configuration

Figures 4-11, 4-12, and 4-13 present three views of the spacecraft as it would be stowed to fit inside of the shuttle payload bay. Figures 4-11 and 4-12 indicate that length is not a problem since the payload bay is about 18-m long and the SRR spacecraft stowed configuration is only about 10-m long. Figure 4-12 shows that there are two very slight interferences with the shuttle in terms of the diameter of the stowed spacecraft, but these are easily fixed as mentioned below. The design of shuttle cradles to support the spacecraft in the payload bay was beyond the scope of this study, although their mass was estimated in a previous study (Reference 3-2) and is included in Table 4-11.

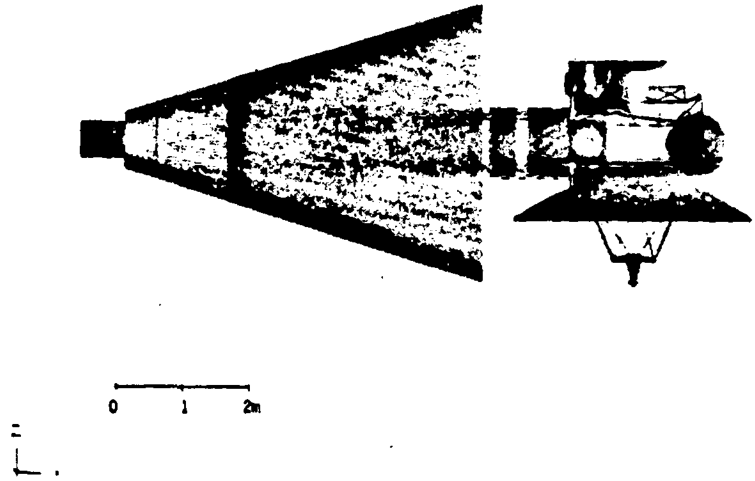

Figure 4-11. Side View of the Stowed Configuration

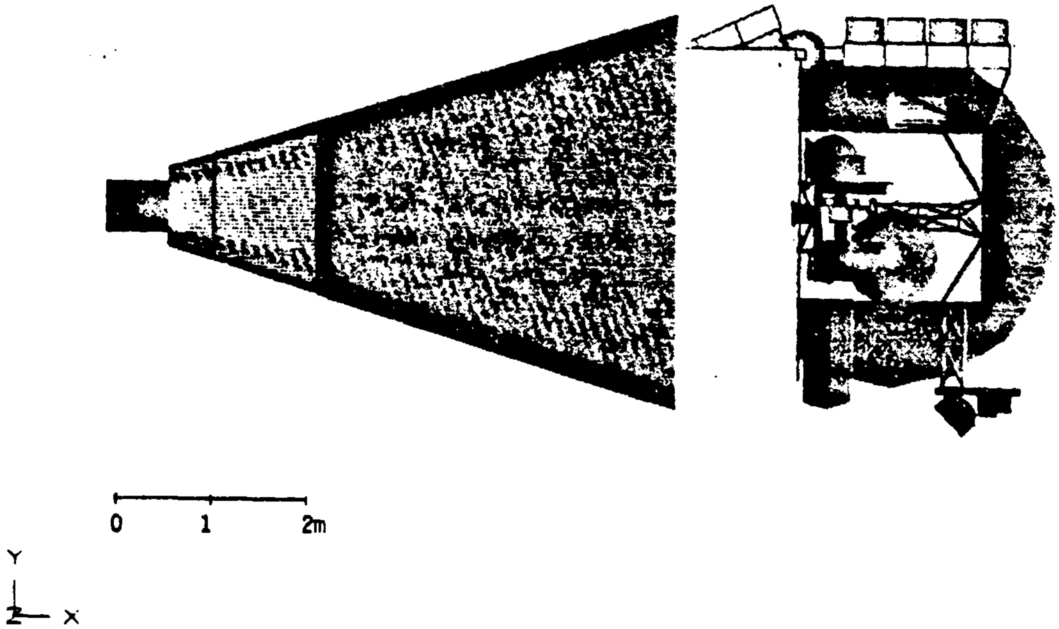

Figure 4-12. Bottom View of the Stowed Configuration

4.7.6 Configuration Problems

The mercury tank provides a shadow of 4.6-m diameter at the bus. The tank was originally 6.7-m closer to the reactor and provided a 5.9-m diameter shadow, but was moved to the present location once the model weights were put into the computer. The tank diameter might be increased to give

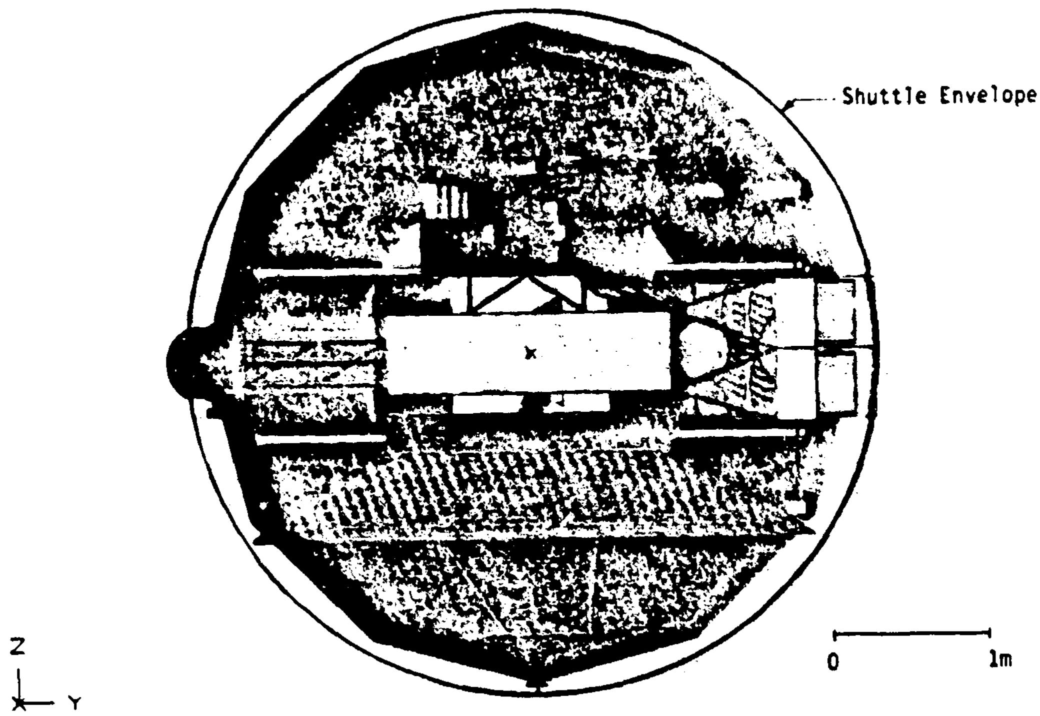

Figure 4-13. Stowed Configuration as Viewed from The Payload End

additional shadow area or the science moved closer to the bus. The intent was that all science would be in the mercury tank shadow with the exception of the magnetometer and the antenna, which are both too long.

The configuration model has several interferences that can be readily fixed. The ion thruster array extends too far outboard radially and would hit the STS walls. In addition, it cuts into the rear attitude control tank. This can be fixed by lowering both thruster arrays slightly and moving the stowed forward array forward and down. The forward array support arm interferes with the interior of the radiator and needs adjusting. The spinning science extends out too far.

The configuration shown does not include a radar experiment.

Finally, the system utilizes the hollow rear volume inside the radiator. Dependent upon the details of the specific SP-100 selected, the stowed configuration must be reviewed for interferences with the SP-100 in this area inside the radiator.

4.7.8 Mass Summary Discussion

The mass of the SP-100 power subsystem (3000 kg) shown in Table 4-11 is the mass goal for the project. The power subsystem component masses are generic estimates and are not meant to represent any one concept.

Depending upon which power subsystem concept, described in Subsection 4.1, is developed and the success of that development, the SP-100 power subsystem could be several hundred kilograms more or less than the 3000-kg goal.

The thrust module contains all the mass of the electric propulsion system, 2145 kg, shown in Table 4-2. The mercury propellant represents the requirement (10,086 kg) plus 300 kg for mercury hold-up and residual for either the basic mission or the basic mission plus Titan probe. The mass of the hydrazine reaction control system (139 kg, Table 4-10) and 103 kg of hydrazine is included in the thrust module. Finally, the thrust module contains 30 kg of attitude and articulation control hardware shown in Table 4-6.

The payload module is what is now the total spacecraft for the Voyager or Galileo planetary missions. Some of the components of the payload module shown in Table 4-11 are modeled after the Galileo spacecraft. The structure shown in Table 4-11 is simply the payload module bus surface area (11 m^2) times the density of aluminum (6.7 g/cm^2, see Subsection 5.2) required to provide shielding from the natural radiation environment. Therefore, the 737-kg of structure is a very simple estimate which, however, is sufficient for the purposes of this study. The masses for mechanical devices, thermal control, harness, pyro, and command and control are taken directly from the Galileo spacecraft. The mass estimate for data storage is twice that of the Galileo. The mass of the attitude and articulation control subsystem is the sum of the payload module and IPPACS platform components shown in Table 4-6. The telecommunications and antenna masses come from Table 4-9. The 100-kg mass of the power processing and distribution subsystem is based upon the Galileo mass (42 kg) but is increased significantly to accommodate the 6-kWe radar. The sum of the mass of the science imaging, particles and fields, and remote sensing instruments is the 176 kg shown in Table 3-6. The mass of the Titan probe and support hardware has been reduced to 213 kg to account for a thinner heat shield, as mentioned in Table 3-7.

The sum of all the spacecraft modules (wet) is 17,645 kg. As discussed in Subsection 4.6 and shown in Figure 4-5, the least capable propulsion option (SCS) can deliver 17,645 kg to 900 km assuming an ASE mass of 3030 kg and a full shuttle payload of 29,500 kg. The OMV and the Star motors could deliver 17,645 kg to an altitude well in excess of 1000 km, i.e., into a very "safe" nuclear safe orbit.

The dry spacecraft mass (assuming all the available mercury and hydrazine has been used) after probe release is 7209 kg or 409 kg in excess of the 6800 kg assumed for the mission performance calculations shown earlier in Table 3-10. The additional dry spacecraft mass will cause the total mission time to be perhaps as much as one year in excess of the 10.43- or 11.26-year mission times shown in Table 3-10. There are several places where the mass estimates shown in Table 4-11 may be too large. There is 77 kg of unallocated contingency in the thrust module. Perhaps the component that could most easily have its mass reduced is the payload module bus structure. The structure mass estimate shown in Table 4-11 is based upon the lowest or most conservative allowable radiation dose. Using the maximum allowable radiation dose as described in Subsection 5.2 and Reference 4-10, would have reduced the required structure mass by about 500 kg. The allowable radiation dose that was assumed is the current Galileo spacecraft allowed dose. As recently as two years ago the proton dose for Galileo was a factor of two higher. This factor of two would reduce the structure mass by about 300 kg. The structure mass is very sensitive to the allowable radiation dose, which is not well defined for a mission such as described in this report.

Even if the mass shown in Table 4-11 cannot be reduced, and even if the mass grows larger (as is likely due to the preliminary nature of this study), the mission could still be performed (although with longer mission times) since there is plenty of launch vehicle margin, thruster lifetime margin, and SP-100 full power life margin.

SECTION 5

ENVIRONMENTS

5.1 METEOROID

The spacecraft will have a relatively high probability of experiencing meteoroid impacts while traveling through the asteroid belt. The meteoroid environment (fluence and probability of impact) was calculated for the baseline Saturn Ring Rendezvous trajectory (see Subsection 3.3) using the meteoroid environmental models described in References 5-1 and 5-2. In the NASA meteoroid environment models used, meteoroids whose origins are cometary are treated separately from those with asteroidal origins. The total fluence is simply the sum of the two types of fluence (fluence = flux x time). For a Poisson distribution, the probability of no impacts on x square meters of spacecraft area is given by:

$$P = (e^{-Fx}) \qquad (5\text{-}1)$$

where

F = particles per square meter

x = square meters of spacecraft area

Calculations are presented for meteoroid masses ranging from 10^{-6} to 100 g. The mean asteroidal and cometary relative velocities are 7.2 and 16.1 km/sec, respectively. The mean asteroidal and cometary densities are 3.5 and 0.5 g/cc, respectively. For example, if the fluence F of 10^{-3} g meteoroids were $1.98 \times 10^{-2}/m^2$ and x is set equal to the total spacecraft area, say 10 square meters, then the probability P of not being hit by a meteoroid of mass 10^{-3} g or larger is 0.974. If this is considered the minimum acceptable probability, then the spacecraft surface area should be designed to survive a meteoroid impact of 10^{-3} g at the given mean velocity.

The probability of no impacts on one square meter of spacecraft area for the entire mission for cometary and asteroidal material is shown in Table 5-1. Bear in mind that several U.S. spacecraft have successfully traversed the same regions of space that this mission would take this spacecraft through.

The fluence of cometary and asteroidal particles for each of the mission phases is shown in Tables 5-2, 5-3, and 5-4.

5.2 REACTOR AND NATURAL RADIATION ENVIRONMENTS

This subsection describes the natural radiation environment encountered during the basic 3810-day Saturn Ring Rendezvous mission (see Subsection 3.3) and the effectiveness of using the mercury propellant as a shield for reactor produced gamma and neutron radiation.

Table 5-1. Total Saturn Ring Rendezvous Mission Meteoroid Impact Probability

Mass, g	Probability of no impacts on one m^2	
	Cometary	Asteroidal
10^{-6}	0.0	2.97×10^{-2}
10^{-5}	0.0	0.60
10^{-4}	0.0	0.93
10^{-3}	2.38×10^{-28}	0.989
10^{-2}	2.02×10^{-2}	0.998
10^{-1}	0.7874	0.9998
1	0.9855	0.99997
10	0.9991	0.999995
100	0.999945	0.999999

Table 5-2. Earth Escape 0-425 Days

Mass-M (g)	Cometary Fluence (Particles/m^2 of mass greater than or equal to M)	Asteroidal Fluence (Particles/m^2 of mass greater than or equal to M)
10^{-6}	2.23×10^{0}	0.0
10^{-5}	1.40×10^{-1}	0.0
10^{-4}	8.54×10^{-3}	0.0
10^{-3}	5.23 x $\times 10^{-4}$	0.0
10^{-2}	3.20×10^{-5}	0.0
10^{-1}	1.96×10^{-6}	0.0
1	1.20×10^{-7}	0.0
10	7.34×10^{-9}	0.0
100	4.50×10^{-10}	0.0

Table 5-3. Heliocentric Phase 425-3075 Days

Mass-M (g)	Cometary Fluence (Particles/m^2 of mass greater than or equal to M)	Asteroidal Fluence (Particles/m^2 of mass greater than or equal to M)
10^{-6}	1.66×10^{1}	3.52×10^{0}
10^{-5}	1.01×10^{0}	5.08×10^{-1}
10^{-4}	6.21×10^{-2}	7.34×10^{-2}
10^{-3}	3.80×10^{-3}	1.06×10^{-2}
10^{-2}	2.33×10^{-4}	1.53×10^{-3}
10^{-1}	1.43×10^{-5}	2.22×10^{-4}
1	8.73×10^{-7}	3.21×10^{-5}
10	5.35×10^{-8}	4.63×10^{-6}
100	3.28×10^{-9}	6.70×10^{-7}

Table 5-4. Saturn Spiral Phase 3075-3810 Days

Mass-M (g)	Cometary Fluence (Particles/m^2 of mass greater than or equal to M)	Asteroidal Fluence (Particles/m^2 of mass greater than or equal to M)
10^{-6}	2.77×10^{5}	0.0
10^{-5}	1.70×10^{4}	0.0
10^{-4}	1.04×10^{3}	0.0
10^{-3}	6.36×10^{1}	0.0
10^{-2}	3.90×10^{0}	0.0
10^{-1}	2.39×10^{-1}	0.0
1	1.46×10^{-2}	0.0
10	8.95×10^{-4}	0.0
100	5.48×10^{-5}	0.0

5.2.1 Van Allen Belt Electron and Proton Fluence Spectra During Earth Escape

The trapped electron and proton fluence spectra have been determined for the Earth escape spiral phase of the SP-100 NEP Saturn Ring Rendezvous Mission. References 4-10 and 5-3 provide the average daily fluences a spacecraft would encounter in a circular orbit as a function of charged particle energy, orbit inclination, and orbit altitude. Using the information in Subsection 3.3 and References 4-10 and 5.3 for particle fluence vs altitude profiles, the daily fluences at <u>0 degrees inclination</u> was integrated for the first 300 days of the spiral up to 33,000 km. At the end of this subsection, a correction factor is applied for 30° inclination, which is more accurate for this mission. The Earth's radiation belt fluences are significantly lower beyond 33,000 km and can be considered negligible.

Table 5-5 presents the calculated integral omnidirectional fluence spectra that was obtained from the above referenced data base. Using these spectra as an external charged particle source, a transport calculation was made through a spherical and double slab aluminum shield using the ADJOINT Monte Carlo Program for electrons and protons. Partial results of this analysis are shown in the dose/depth and the proton fluences curves of Figure 5-1. It is noted that at 0.1 g/cm^2 aluminum (15 mils) electrons and protons are of equal importance and after this thickness the protons yield the most dose. The proton fluence is the 20-MeV equivalent displacement damage fluence to be used in the spacecraft calculations.

5.2.2 Saturn Electron Fluence Spectra

The trapped electron fluence spectra for the spiral trajectory into Saturn has been determined from work reported in Reference 5-4. From this work, the integral electron fluence was calculated using the Saturn trajectory reported in Subsection 3.3. The calculated integral electron fluence is shown in Table 5-6. It is noted that no proton fluence spectra was calculated because the proton high energy response was barely above the background noise.

The radiation transport of this spectra through various thicknesses of aluminum is shown in Figure 5-2.

5.2.3 Free Space Solar Flare Protons

The solar flare proton fluence spectra in this analysis was obtained from Reference 5-5. This reference reports a model of an anomalously large (A.L.) solar event of August 1972. The radiation transport results of this model through aluminum is shown in Figure 5-2. For a 10-year mission, the number of these events that should be considered, for a 90% confidence level that the proton fluence will not be exceeded, is approximately seven events. The latter number was extrapolated from Reference 5-3. It is noted that electrons dominate up to about 9 g/cm^2 of aluminum thickness. Considering all species of radiation shown in Figures 5-1 and 5-2, the protons derived from spiraling through the Van Allen belts will yield the most dose. However, the spacecraft shield design will be based on the total dose and total proton fluence of all species as shown in Figure 5-3.

Table 5-5. Integral Fluence Spectra for Earth Escape

Energy (MeV) E	Integral Electron Fluence Electrons/cm^2 (>E)
0.1	4.3 + 15
0.5	1.2 + 14
1.0	2.6 + 13
2.0	3.5 + 12
3.0	4.0 + 11
4.0	1.6 + 10
Energy (MeV) E	**Integral Proton Fluence Protons/cm^2 (>E)**
0.1	2.1 + 15
0.3	8.0 + 14
1.0	2.0 + 14
4.0	1.6 + 13
10	3.7 + 12
30	6.0 + 11
100	7.5 + 10
400	3.7 + 9

5.2.4 Uncertainty Factor Recommended for Study

The uncertainty in the particle intensities used in the analysis is largely due to their variations with time and position. The calculated fluences are averaged over a wide range of time and altitudes and, therefore, may be taken to have a more modest uncertainty of around a factor of two. This is the uncertainty factor recommendation for this study.

Table 5-6. Integral Electron Fluence Spectra for Saturn Rendezvous

Energy (MeV) E	Integral Electron Fluence Electrons/cm^2 (>E)
0.1	6.3 + 13
0.3	5.5 + 13
1.0	2.2 + 13
3.0	5.8 + 12
10	4.8 + 11
31	5.5 + 10

Integral Solar Flare Proton Spectra (From Reference 5-5)

Energy (MeV) E	Integral Proton Fluence Protons/cm^2/(1) Anomalously Large (A.L.) Event
10	1.680 + 10
20	1.152 + 10
30	7.900 + 9
40	5.417 + 9
50	3.714 + 9
60	2.547 + 9
70	1.746 + 9
80	1.197 + 9
90	8.210 + 8
199	5.629 + 8

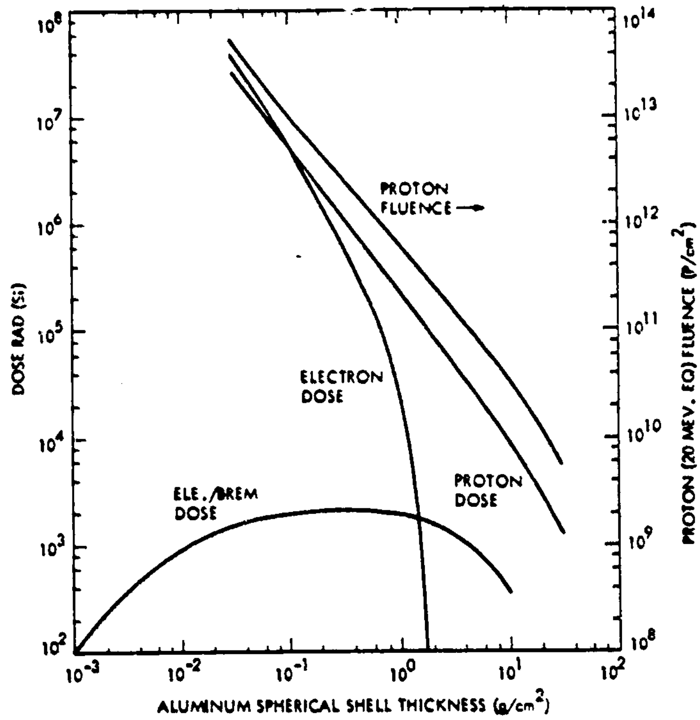

Figure 5-1. Dose/Depth in Aluminum of the Earth Spiral Escape (Zero Degrees Inclination) for the Nominal Saturn Ring Rendezvous Mission

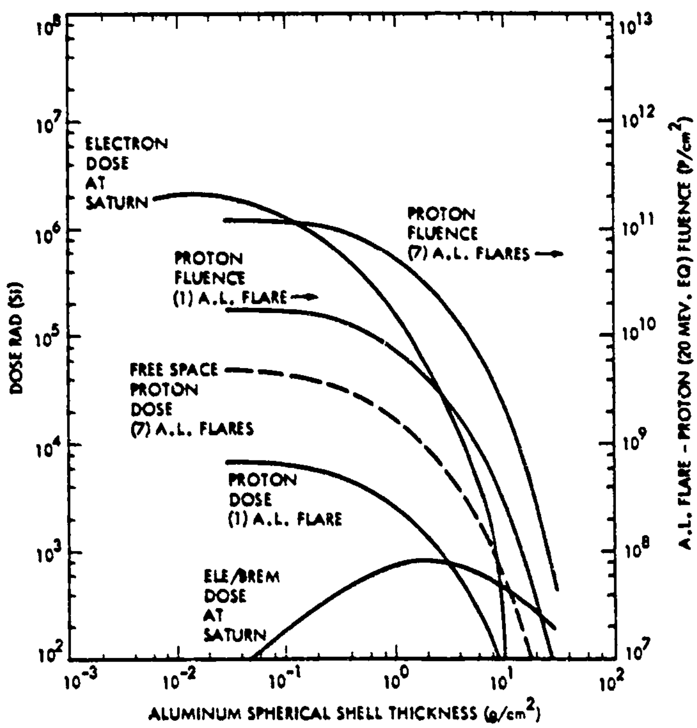

Figure 5-2. Dose/Depth in Aluminum of the Heliocentric and Saturn Spiral Phases of the Nominal Saturn Ring Rendezvous Mission

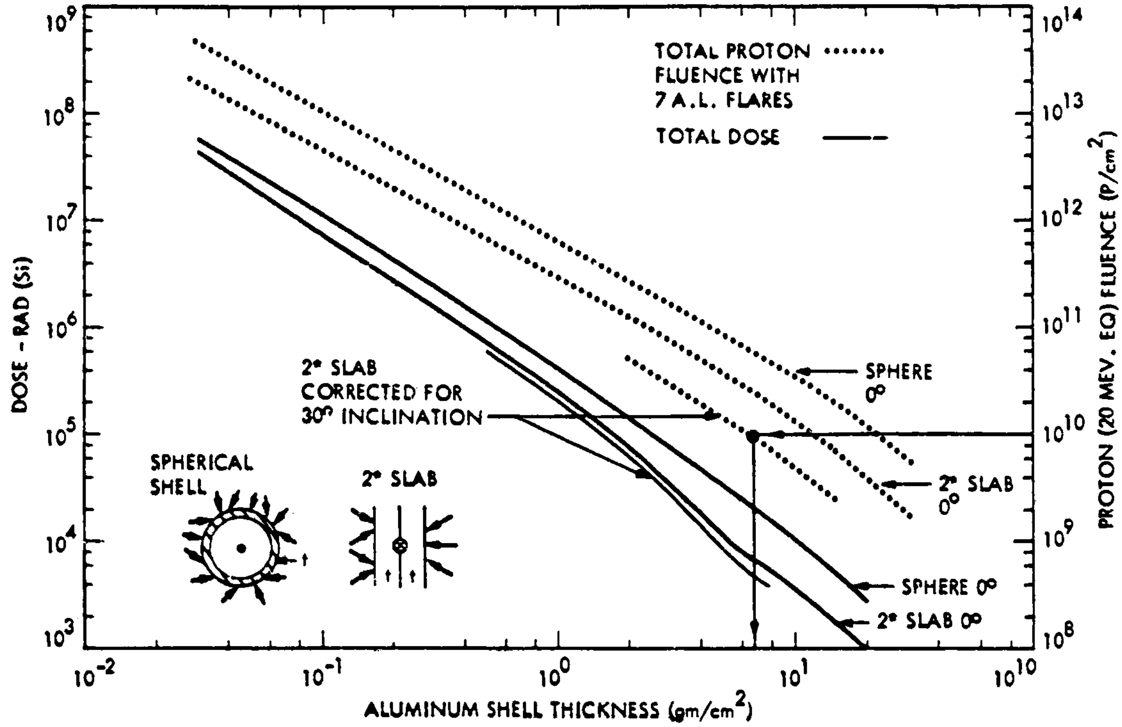

Figure 5-3. Dose/Depth in Aluminum for the Total Saturn Ring Rendezvous Mission

5.2.5 Reactor Radiation Attenuation by Mercury

The intrinsic worth of the mercury propellant as a radiation shield was investigated. As discussed in Subsection 4.4, the mercury propellant tank was positioned of the centerline of the spacecraft and configured into a disk 2.2 m in diameter and about 0.3-m thick. As discussed, the tank was loaded with approximately 10,000 kg of mercury.

As discussed in Subsection 4.1, the SP-100 radiation dose after seven years of full power operation at a dose plane 25 m from the reactor is required to be 1 x 10^{13} neutrons/cm^2 (1 MeV) and 5 x 10^5 Rads (Si). The reactor full power operating time for the basic Saturn Ring Rendezvous mission (see Subsection 3.3) is 5.85 years. During this mission, the mercury depth in the propellant tank would be depleted as is shown in Figure 5-4. Obviously, the worth of the mercury shield is much greater at the start of the mission that at the end; however, on an integrated basis, it is still quite effective. The integrated neutron fluence and gamma dose at 25 m from the reactor at the end of the mission were computed using the mercury depletion profile shown in Figure 5-4. The results show that the neutron fluence was reduced by about 74% to 2.6 x 10^{12} neutron/cm^2 (1 MeV) and the gamma dose reduced by about a factor of 20 to 2.5 x 10^4 Rads (Si).

5.2.6 Radiation Design Limits and Payload Placement

Based upon recent JPL experience in the design of radiation hard planetary science spacecraft and the trends in electronic part hardness,

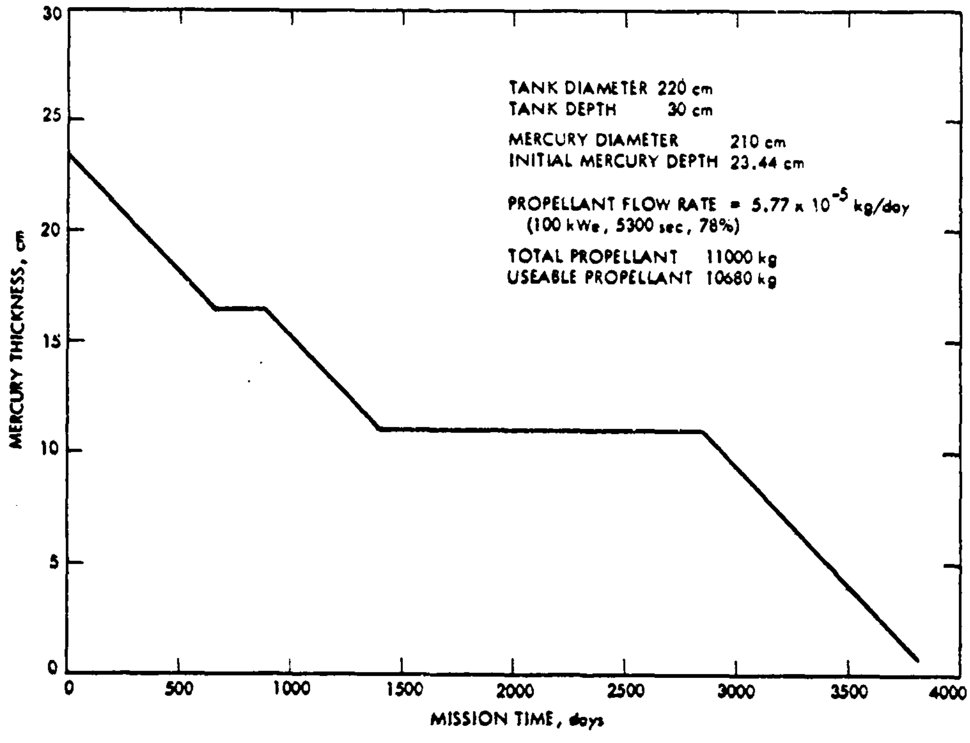

Figure 5-4. Mercury Depth Versus Mission Time

Reference 5-6 recommended the following two levels of radiation dose for design purposes.

(1) Medium Environment (Similar to Galileo)

(a) Total dose 75 kRads inside the spacecraft bus.

(b) Neutron fluence 1×10^{12} N/cm^2 1 MeV equivalent.

(c) Proton fluence 1×10^{10} P/cm^2 20 MeV equivalent.

(2) Maximum Environment

(a) Total dose 250 kRads inside the spacecraft bus.

(b) Neutron fluence 1×10^{13} N/cm^2 1 MeV equivalent.

(c) Proton fluence 1×10^{11} P/cm^2 20 MeV equivalent.

Both environments were selected with a total dose radiation design margin (RDM) of 2 for all semiconductor devices. That is, an environment of 75 kRad (Si) with a RDM of 2 would require all semiconductor devices to be rated at 150 kRad (Si).

Table 5-7 presents the distance at which the radiation from the SP-100 with and without the mercury tank, reaches the limits of the two recommended environments.

Table 5-7. Distance from SP-100 for Minimum Environment (m)

Power Subsystem Comparison	Environment			
	Medium		Maximum	
	Gamma	Neutron	Gamma	Neutron
Basic SP-100	64.5	79	35.4	25
SP-100 with Mercury Shield for Saturn Ring Rendezvous Mission	*	40.3	*	*

*The distance is less than 20 m.

It was desired to use the medium environment so as to be conservative from an electronic parts point of view. As discussed in Subsection 4.4 the payload was placed at 65 m from the reactor since this distance meets the gamma limit for the medium environment and, with the mercury shield, allowed a large margin for radiation dose from the natural environment. At 65 m, the gamma dose and neutron fluence behind the mercury shield are 3.7 kRad (Si) and 3.85×10^{11} neutrons/cm^2.

Using the medium environment as the design criteria, one may estimate the required thickness of aluminum wall thickness for the spacecraft bus. Figure 5-3 is used to make this estimate. Figure 5-3 shows dose and fluence as a function of aluminum thickness for both a spherical shell and double slab approximations. A single slab approximation (not shown) would allow half the dose or fluence at a given aluminum thickness than the dose from the double slab. Various locations within the spacecraft bus will look like the spherical shell, double slab, and single slab approximations. For studies such as this, the double slab approximation is used in an attempt to obtain an average solution for the entire spacecraft.

Figure 5-3 also shows the corrections made to the dose and fluence double slab approximations in order to account for a trajectory that maintains a ~30° (28.5°) inclination with respect to the Earth's equatorial plane during the Earth spiral escape portion of the mission. The Earth's natural radiation environment is lower by a nontrivial amount at 30° compared to 0° inclination. An estimate of this difference for a NEP spiral escape was made based upon the results obtained for a 30° trajectory presented in Reference 3-2. The proton environment at 30° is roughly a factor of 3 or more less than that at 0°. The electron environment at 30° is roughly a factor of 1.5 less than that at 0°. These factors have been used in the corrected lines for 30° on Figure 5-3. The total dose and fluence does not decrease by a factor of 3 and 1.5 due to the contributions from the free space and Saturn environments.

Using the medium environment criteria, one can see that the proton fluence environment is the most stressing and leads to a required aluminum wall thickness of about 6.7 g/cm^2. The combined natural and reactor radiation environment at the payload, inside of the spacecraft bus (shielded at 6.7 g/cm^2) over the 3810-day nominal mission is: 1 x 10^{10} protons/cm^2 (20 MeV equivalent), 8.5 kRad (Si) where 3.7 kRad is from the SP-100 and 4.8 is from the natural environment, and finally 3.85 x 10^{11} neutrons/cm^2 (1 MeV equivalent).

5.3 SATURN'S RINGS

A general description of Saturn's rings and moonlets is presented in Table 5-8 (from References 5-7 and 5-8). In general, the main rings (A-D) are thin (less than about 200 m) and are populated by relatively large objects 0.01 to 5 m in diameter. Objects as large as several kilometers are rare, but probably do exist. The larger objects lie in a monolayer in the ring plane while the smaller particles account for the vertical depth of the rings. Large scale waves due to gravitational perturbations are thought to take place in the main rings with a vertical amplitude of about 1 km. The outer rings (E, F, and G) appear to be quite different than the main rings. The E- and G-rings are made up almost entirely of micron size dust, but they are at least 1000-km thick. The F-ring is very thin radially (10s of kilometers).

As discussed in Subsection 3.3.3, the trajectory for the Saturn Ring Rendezvous mission lies in the ring plane until the spacecraft reaches the G-ring (~2.8 R_S). Starting at the G-ring, the spacecraft follows the non-Keplerian orbit which rises 18 km above the ring plane at the inner edge of the D-ring. Therefore the trajectory takes the spacecraft <u>through</u> the E-, G-, and maybe F-rings and <u>above</u> the A-D rings (see Figure 3-5). The hazard associated with this trajectory will be greatest in the E-ring due to its large size. Although the fluence will be very large (between 4 x 10^7 and 4 x 10^6 particles of all sizes per square meter), the relative velocity between the spacecraft and the dust is very low (~10s m/sec) (see Subsection 3.3.3). It has also been suggested that the spacecraft be oriented with the long spacecraft axis parallel to the relative velocity vector with the SP-100 power system facing "into the wind." This is probably a good idea since the SP-100 power system is very hard to dust, especially compared to the science payload, and therefore can, to some extent, shield the science payload from the E-ring dust. This orientation, however, would require a different configuration from that shown in Subsection 4.7.

Table 5-8. Saturn's Rings and Moonlets

R_S = Saturn Radius = 60,330 km
M_S = Saturn Mass = 5.685×10^{29} g

Feature	Location (R_S)	Ring Thickness	Mass (M_S)	Particle Size
D-Ring	1.11-1.235	10 m	---	0.01 m
C-Ring	1.235-1.525		2×10^{-9}	
B-Ring	1.525-1949	to	5×10^{-8}	to
Cassini Division	1.949-2.025		1×10^{-9}	
A-Ring	2.025-2.267	200 m	1.1×10^{-8}	5 m
1980S28 Moonlet	2.282	---	1.47×10^{-11}	---
1980S27	2.31	---	1.03×10^{-9}	---
F-Ring	2.324	?	---	?
1980S26	2.349	---	---	---
1980S3	2.510	---	---	---
1980S1	2.511	---	---	---
G-Ring	2.82	1000 km	1.4×10^{-17}	Dust: $0.1\text{-}10 \times 10^{-6}$ m
Mimas	3.075	---	---	---
E-Ring	3.0-8.0	1000-2000 km	---	Dust: $0.1\text{-}10 \times 10^{-6}$ m
Enceladus	3.946	---	1.48×10^{-7}	---

SECTION 6

SATURN RING RENDEZVOUS MISSION COSTS

All estimates have been generated using the Science Applications International Corporation (SAIC) cost models as appropriate. Hardware data are entered into the models from the JPL systems definitions for the Mission Module and Nuclear Electric Propulsion system, and from Ames Research Center specifications for a Titan Probe. Given the preliminary level of definition and advanced technology requirements for the SRRPR mission, a 30% contingency has been applied to the cost estimates. All estimates are presented in FY 1984 constant dollars.

6.1 SCOPE

The four SRR mission options, which have been costed, are as follows:

(1) Baseline SRR plus radar (SRRPR).

(2) SRRPR + Titan probe.

(3) SRRPR + Titan radar mapping.

(4) SRRPR + Titan probe + Titan radar mapping.

To a first order approximation, the study team assumed that Options 1 and 3 would have identical development project costs (i.e., costs through launch + 30 days) and that Options 2 and 4 would have identical development costs. Conversely, flight project costs have been separately estimated for each option. Details of the development and flight project cost estimates are presented in Tables 6-1 and 6-2, respectively.

6.2 DEVELOPMENT PROJECT COSTS

Definitions of the Mission Module science and engineering hardware were taken from References 6-1, 6-2, and 6-3 and used as inputs to the Planetary Program cost model in Reference 6-4. Significant cost drivers include the 6-kWe radar, the 440-W X-band transmitter, and the spacecraft power subsystem. Also, the estimates (References 6-1 and 6-2) for low design heritage contribute to relatively high cost estimates for both attitude and articulation control and command and data handling. A probe spin table and relay radio have been added to the Option 2/4 Mission Module. Finally, cost of the EP/Payload Boom is book-kept with the Mission Module.

For costing the Nuclear Electric Propulsion subsystem, the team used selected algorithms from the Solar Electric Low-Thrust System Cost Model in Reference 6-5. In practice, the team simply ignored those algorithms which deal with the solar array. The team assumed that the propulsion subsystem has already been developed for some prior application; thus, the cost estimate

Table 6-1. Saturn Ring Rendezvous Development Cost Summary (FY 84 $M)

	Options 1 & 3		Options 2 & 4	
Mission Module		386		398
Project Management	27		28	
Science	100		100	
Engineering	259		270	
Titan Probe				151
Project Management			9	
Science			57	
Engineering			60	
G&A* and Fee (20%)			25	
Propulsion System		92		92
Project Management	8		8	
Engineering	69		69	
G&A* and Fee (20%)	15		15	
Power System		45		45
Systems Integration		21		24
Mission Design		20		25
Contract Monitor		7		19
Launch + 30^d Ops		35		40
Program Management		18		24
Net Total		624		818
Contingency (30%)		187		245
Grand Total		811		1063

*G&A = General and administrative costs.

Table 6-2. Saturn Ring Rendezvous Mission Cost Summary (FY 84 $M)

	Option 1	Option 2	Option 3	Option 4
Development Project	811	1063	811	1063
Flight Project				
Flight Operations	59	63	67	67
Data Analysis	25	27	30	35
Program Management	3	3	3	3
Contingency (30%)	26	28	30	32
Grand Total	924	1184	941	1200

presented here is essentially a recurring (unit) cost except for some nonrecurring costs associated with EP/Payload interfaces.

For the SP-100 power subsystem, the team has merely taken the midpoint of the supplied unit cost range of $20M to $70M.

Definitions for a Galileo-derivative Titan Probe were taken from work performed for NASA Ames Research Center (ARC). During the study, ARC calculated an entry shield mass of 24 kg. Since the SRRPR mission would occur significantly later than ARC's Titan Flyby/Probe mission, the team had to relax its previous assumptions regarding Probe system heritage, leading to a more conservative cost estimate than that shown in Reference 6-6 ($109M, FY 84 dollars). Finally, the Probe includes preentry science with suitable attachments and interfaces.

The team has assumed that the Mission Module would be developed in-house with all other systems developed or bought through system contracts. Thus, the team applied 20% to the propulsion subsystem and probe for contractor general and administrative (G&A) costs and fee. The team presumes these are already included in the $45M power subsystem cost. The remaining items in Table 6-1 are self-explanatory.

6.3 FLIGHT PROJECT COSTS

Flight operations costs were estimated by applying a set of fixed rates to the various phases of the mission flight project for each option. The phases and times for each option were taken from Subsection 3.3.4, and are summarized in Table 6-3.

Table 6-3. Flight Phase and Time

Phase	Total Phase Duration, months			
	Option 1	Option 2	Option 3	Option 4
Cruise Coast	55.0	64.9	64.9	62.8
Cruise Thrust	49.3	49.3	49.4	51.7
Encounter Thrust	20.8	20.9	23.5	23.5
Encounter Coast	---	---	2.0	2.0
Total	125.2	135.1	139.8	140.0

Data analysis costs are estimated based on the total science complement and nominal encounter duration for each option.

6.4 SRRPR BASELINE SENSITIVITY

Figure 6-1 shows the sensitivity of the SRR baseline mission development cost to uncertainty in the power subsystem unit cost. The study team assumed that such a large uncertainty in cost indicates significant technical uncertainties, which will impact all levels of integration (e.g., power subsystem to propulsion subsystem, then to Mission Module, and finally to launch stack). The cost impact is linear with a ratio of approximately 1.8:1.

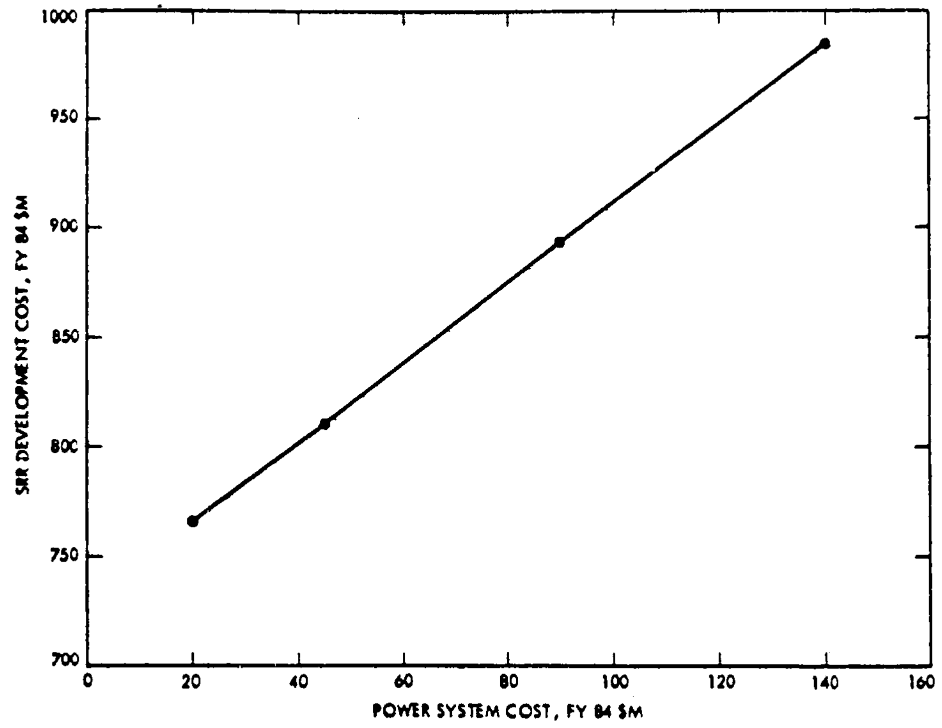

Figure 6-1. The Sensitivity of the Nominal Saturn Ring Rendezvous Mission Development Cost to the Recurring Cost of the SP-100 Power Subsystem

SECTION 7

NEP MISSION PLANS

7.1 BACKGROUND

The Solar System Exploration Committee (SSEC) published its recommended Core Program for planetary exploration in May, 1983 (Reference 2-11). The missions in the Core Program were selected and defined with the ground rule that they would not require expensive new technologies such as nuclear electric propulsion (NEP). At that time, the SSEC also recommended that the Core Program be augmented at the earliest opportunity. Augmentation is understood to mean a significant advancement in exploration, which involves intensive, high technology missions to study individual bodies or planet systems having identified scientific interest and priority. Such missions are difficult to perform, generally require advanced propulsion or spacecraft configurations, may have multielement payloads, and are costly compared to Core Program missions.

The SSEC continued its deliberations and agreed to provide, in a second report, detailed recommendations of mission activities suitable for the Augmentation Program. Several high-priority planetary science objectives were already identified by the Space Science Board's Committee on Planetary and Lunar Exploration (COMPLEX) (see Section 2). These include the operations of a mobile scientific rover on the surface of Mars, the automated collection and return of samples from that planet, and the return to Earth of samples from comets and asteroids. Other mission candidates could be gleaned from the accepted strategy of a balanced approach to solar system exploration. In particular, COMPLEX was in the process of updating its strategy for outer planet exploration. The COMPLEX report and the SSEC Augmentation Program report are due to be published in early 1986.

7.2 MISSION CANDIDATES

Table 7-1 lists a wide variety of example augmentation plan mission candidates across the spectrum of inner planets, primitive bodies, and outer planet targets. None of these missions are currently included in the Core Program, although it is possible that certain "moderate augmentation" missions could be incorporated into an extended Core Program definition. Examples in this category might include Mercury Orbiter, Jupiter Atmosphere Multiprobes, Galileo Satellite Penetrators, and, possibly, the upgrading of the Uranus and Neptune flyby/atmosphere probes to orbiter status. Of those missions considered to require major augmentation, the Mars Rover/Sample Return and the Comet Nucleus Sample Return have been clearly identified by the SSEC as high-priority candidates. No specific mission candidate for the outer planets has been selected. The reasons for this deferral were the large number and diversity of outer planet worlds, the wide interest of the science community, and the data yet forthcoming from the Galileo and Core Program missions needed to wisely plan intensive investigations. Also awaited was the outer planet strategy being deliberated by COMPLEX. <u>Intensive study of the Saturn system has since been identified by COMPLEX as the highest priority objective of outer planet exploration</u>. Important objectives for the Jupiter system,

Table 7-1. Example Augmentation Plan Mission Candidates

• Inner Planets	
	Mars Rover/Sample Return*
	Venus Long-Lived Lander
	Venus Sample Return
	Mercury Orbiter/Lander
• Primitive Bodies	
	Comet Nucleus Sample Return*
	Multiple Asteroid Sample Return
• Outer Planets#	
	Jupiter Atmosphere Multiprobes
	Galilean Satellite Penetrators
	Europa Orbiter/Lander
	Io Lander/Rover
	Saturn Ring Rendezvous
	Titan Orbiter/Buoyant Stations
	Titan Lander
	Uranus Orbiter/Probe
	Neptune Orbiter/Probe
	Triton Lander
	Pluto Orbiter
	Pluto/Charon Lander

*SSEC-Identified priority.

#Intensive study of Saturn system identified by COMPLEX as highest priority objective for outer planets exploration.

particularly the Galilean satellites, have also been expressed in terms of augmentation missions.

Figure 7-1 is included here to give some perspective of the mass requirements for the class of example augmentation mission candidates when implemented by the usual ballistic flight mode, i.e., chemical propulsion. Note that the least capable injection stage here is the Centaur G' with a single shuttle launch; this was the most capable launcher assumed for Core Program missions. The size of the mission requirement rectangles, while partly subjective, does encompass realistic, design trade-off options. Among the factors of variability, depending upon the mission, are launch year opportunity, propulsive orbit capture vs. aerocapture, Earth-storable vs. space storable retropropulsion, and target body selection (in the case of comets and satellites). It is quite clear that the Shuttle/Centaur G' has very limited application for these difficult missions. On-orbit fueling and/or assembly of large upper stages may be needed for ballistic mode implementation. Apart from the cost of multiple shuttle launches, this requirement is not necessarily detrimental since the technology is likely to be available in the Space Station era.

7.3 AN EXAMPLE AUGMENTATION PROGRAM PLAN

Taking direction from the SSEC and COMPLEX recommendations as discussed earlier, an attempt is made to fashion an augmentation program plan that is balanced, not too overly ambitious, and is "spread out" over a reasonable period of time with first launch not before the mid-1990s. The following set of four missions has been selected as an example:

(1) Mars Sample Return (MSR) ....1996 launch

(2) Europa Orbiter/Penetrators (EOp) ....1999 launch

(3) Comet Nucleus Sample Return (CNSR) ....2002 launch

(4) Saturn Ring Rendezvous (SRR) ....2005 launch

Figure 7-2 shows the timelines for these missions beginning with project start date and ending with science data return. Note the NEP is employed in this plan only for the last two missions in the set, although, if available, NEP could be used on the other missions as well.

The Mars Sample Return mission is based on the reference concept (chemical propulsion) selected at the end of the FY 84 joint study activity.* It employs the techniques of an out-of-orbit entry and Mars orbit rendezvous, and assumes aerocapture/aeromaneuvering technology. Injected mass requirements are easily satisfied by the fully-fueled Centaur G' (Curve 4 in Figure 7-1), but the Shuttle/Centaur G' might be apropos if further mass reduction can be shown in FY 85 studies. The Europa Orbiter/Penetrators

*NEP has been suggested as an option for Mars Sample Return and is being included in the FY 85 study.

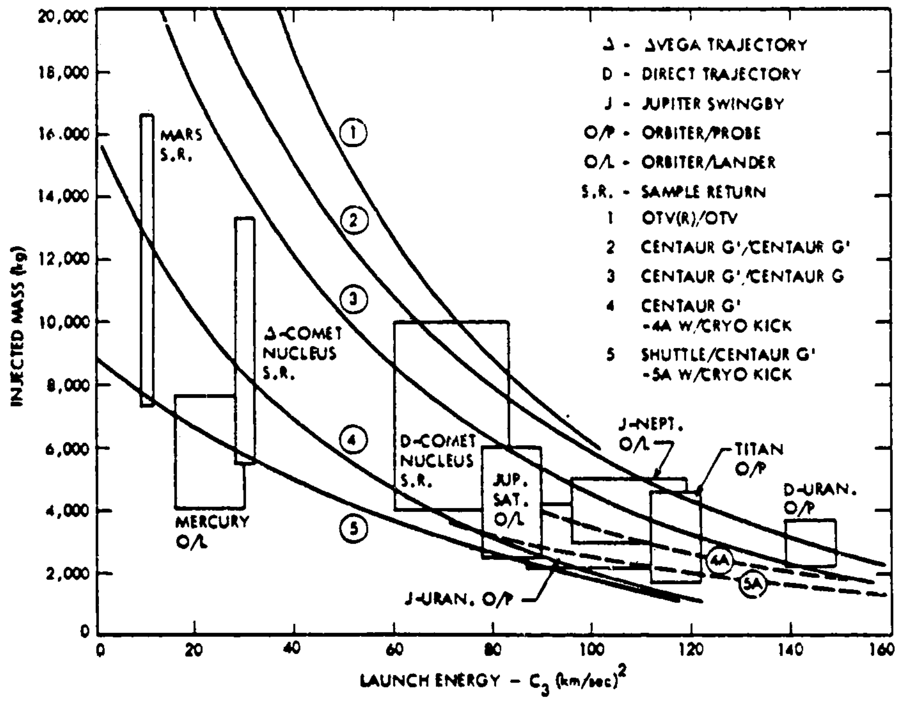

Figure 7-1. The Performance of Various Upper Stages with On-Orbit Assembly or Fueling for Potential Planetary Missions (Ballistic Missions)

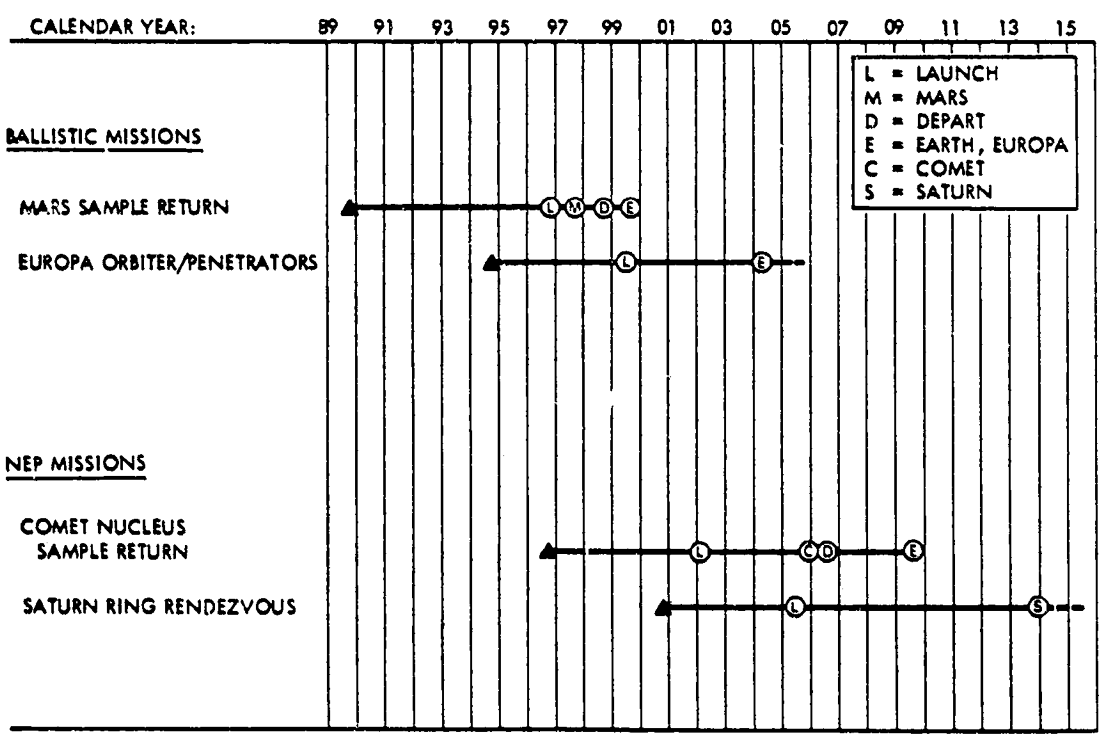

Figure 7-2. Mission Timelines for the Example Augmentation Mission Plan

mission is based on a study by Science Applications International Corporation (SAIC) several years ago. It is launched by the Shuttle/Centaur G' and employs the ΔVEGA flight mode with Earth-storable retropropulsion. A special gravity-assist tour between the Callisto-Ganymede and then the Ganymede-Europa satellite pairs provides minimization of Europa orbit insertion requirements. The spacecraft carries three penetrator systems (including retro), which could be deployed singly at Callisto, Ganymede, and Europa, or, possibly, all three at Europa.

NEP application for both the comet and Saturn missions assumes a single shuttle launch to a 700-km altitude orbit and the Earth escape spiral mode. In the example plan, Wild 2 is the target of the Comet Nucleus Sample Return mission; Kopff could also be the target body if the launch date is modified by less than one year. Comet rendezvous and sampling operations take place in the vicinity of aphelion for the purpose of minimizing the dust hazard. Total round trip time is about 7.5 years with a 100-kWe NEP design employing 10 thrusters (4 operating with spares switched in to satisfy thruster lifetime constraints). The Saturn Ring Rendezvous is the reference mission discussed in Subsection 3.3 and costed as a stand-alone mission in Section 6.

Table 7-2 shows estimated development costs for each of the four missions in the augmentation plan. These costs range from $636 to $2175M in constant FY 84 dollars, including a liberal contingency of 30%. Flight operations and data analysis costs are not included in these estimates; they would add at least another $100M. Note that the estimated development cost for the Saturn mission is less than the $811M stand-alone cost reported in Section 6. The reason for this is mission module inheritance taken from the preceeding Comet Nucleus Sample Return development project.

Figure 7-3 shows the annual funding spreads for the example augmentation program plan. Project new start milestones are assumed to be FY 90 for MSR, FY 95 for EO_p, FY 97 for CNSR, and FY 2001 for SRR. Mars Sample Return is a seven-year development project necessitated by its complexity in hardware development and integration. Annual funding reaches $450M in the peak years. This drops to a level of about $350M during the CNSR peak development years and then to the $200M level for SRR.

Figures 7-2 and 7-3 and Table 7-2 describe an example mission plan that includes NEP but is constrained to be as consistent as possible with the current interpretation of NASA's and the science community's desires for an augmentation program. As such it represents a reasonable projection of the future based upon the current fiscal environment and mission priority viewpoint. Figures 7-4 and 7-5 and Table 7-3 present a NEP planetary mission plan that was developed by taking a "clean piece of paper" and envisioning a future that would capitalize on the capability of NEP and was not severely constrained by the current environment. As such the NEP planetary mission plan represents the most optimistic scenario for NEP implementation on planetary missions. Both plans cover the same time frame and have nearly the same integrated costs. The "NEP plan" includes one more mission than the example augmentation plan for about the same cost, but does so by leaving out the Mars Sample Return mission. Both plans have merits from different points of view. The example augmentation plan is more realistic, but the "NEP plan" provides a useful upper bound on SP-100 NEP planetary missions.

Table 7-2. Example Augmentation Plan Development Costs (FY 84 $M)

	MSR	EOp	CNSR	SRRPR
Spacecraft/Mission Module	282	316	304	270
Entry/Lander Ascent Modules	739		175	
Rover Vehicle	215			
Return Vehicle/Capsule	117		61	
Penetrators		116		
Nuclear Electric Propulsion System	---	---	71	92
Nuclear Electric Power Subsystem	---	---	45	45
System Integration	57	17	22	15
Mission Design	40	25	25	20
Launch + 30^d Operations	71	24	40	26
Contract Monitor	103	9	24	7
Program Management	49	15	23	14
Net Total	1673	522	790	489
Contingency (30%)	502	157	237	147
Grand Total	2175	679	1027	636

7.4 CAVEATS

In the contemporary constrained budgetary environment for planetary exploration, the concept of any plan beyond the Core Program, which includes several missions in sequence, is rather tenuous. By definition, the Core Program is foundational and continuing, while augmentation is special and discrete. It is likely that augmentation funding will require Executive direction based on national goals, or possibly a cooperative endeavor with international partners. It may be that only one or two large missions could be afforded over a ten-year project start interval. On the other hand, it is also possible that the national economic state will improve allowing relaxed fiscal constraints on planetary exploration endeavors. Given this uncertainty, it is probably not wise to place too much stock in any single plan for augmentation missions. Future planning efforts should develop several options for different exploration strategies, based on established scientific priorities, comprising a range of desires (benefits) matched against realistic implementation (costs).

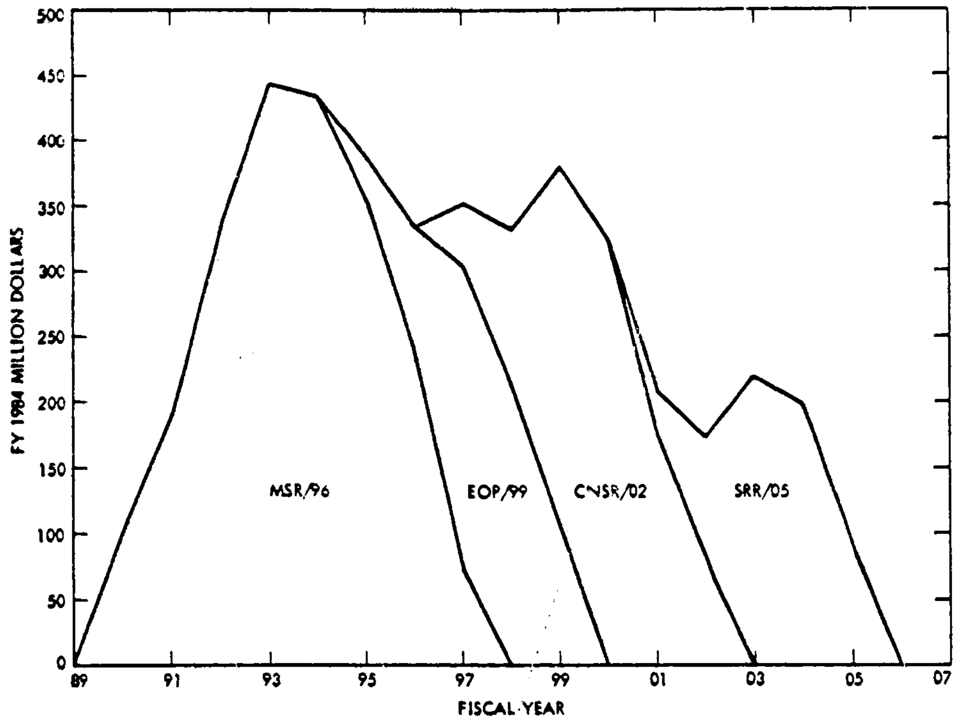

Figure 7-3. Annual Resource Requirements for the Example Augmentation Mission Plan

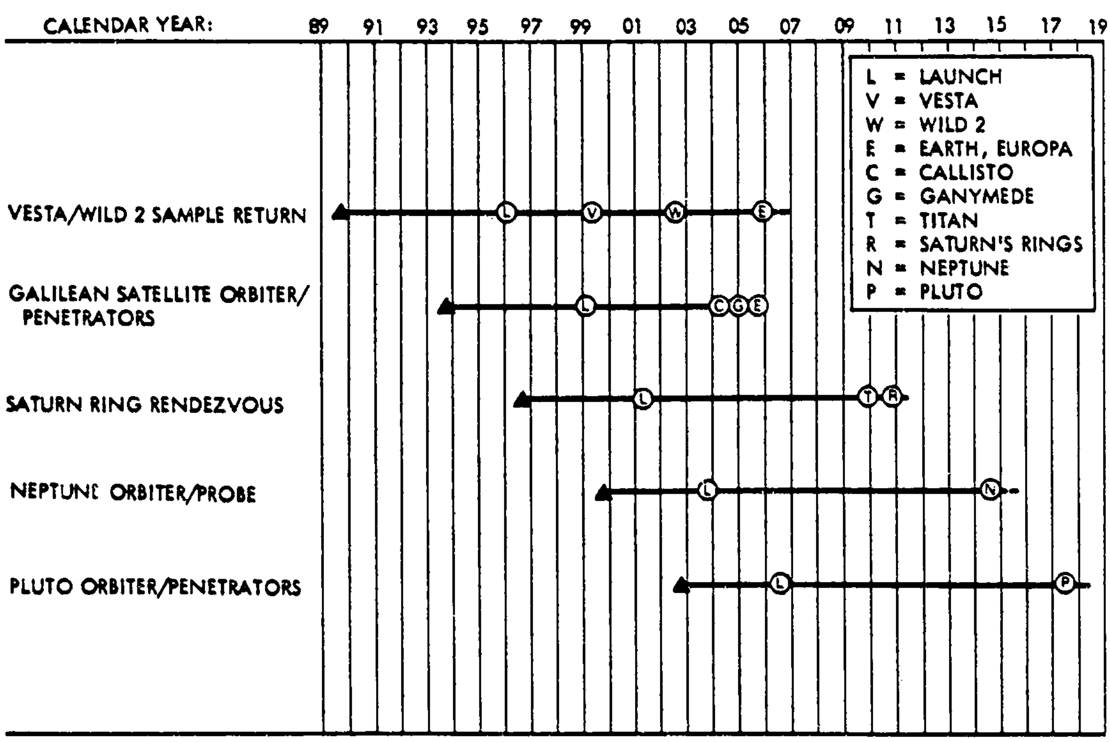

Figure 7-4. Mission Timelines for the NEP Mission Plan

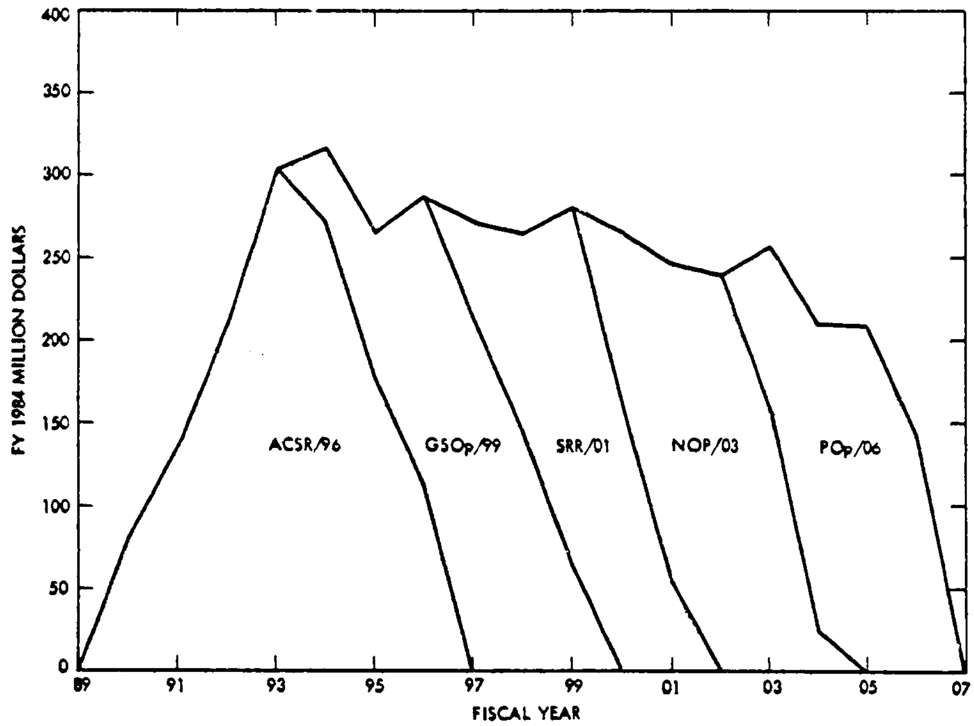

Figure 7-5. Annual Resource Requirements for the NEP Mission Plan

Table 7-3. NEP Missions Plan Development Costs (FY 84 $M)

	ACSR	GSO_p	SRR	NOP	PO_p
Mission Module	304	206	265	206	181
Asteroid Lander/Sampler	158				
Comet Lander	133				
Comet Samplers	53				
Sample Return Capsule	61				
Penetrators		116			96
Probe				126	
Propulsion System	71	92	83	83	83
Power System	45	45	45	45	45
Systems Integration	34	17	14	16	14
Mission Design	25	25	20	20	18
Launch + 30^d Operations	48	30	26	29	26
Contract Monitor	37	16	6	16	14
Program Management	29	16	14	16	14
Net Total	998	563	473	557	491
Contingency (30%)	299	169	142	167	147
Grand Total	1297	732	615	724	638

SECTION 8

REQUIREMENTS ON SP-100

This section contains the primary results of this study, i.e., the requirements on the SP-100 power subsystem originating from the preliminary design of nuclear electric propulsion planetary missions and spacecraft systems. The results presented in this section are constrained by the degree of definition available for both the SP-100 and NEP missions, spacecraft systems, and the time available for this study to investigate those areas of potential requirements. The previous sections of this report as well as References 3-2, 8-1, and 8-2 are the best available definition of NEP missions and spacecraft systems. Reference 8-3 is the best, most concise definition of what the SP-100 power subsystem is currently intended to be.

Requirements on the SP-100 may originate from an interface between or interactions with any of the SP-100 features and NEP mission characteristics or an NEP spacecraft and its subsystems and functions. Tables 8-1, 8-2, and 8-3 presents a summary of the areas of potential interaction from which requirements on the SP-100 power subsystem could originate. Using these tables as a guide for those areas in which to investigate, the SP-100 requirements document (Reference 8-3) was reviewed in order to identify any requirements that needed to be changed, added, or deleted in order that a potential SP-100 power subsystem could be successfully integrated into an NEP planetary mission. The following lines use the structure of Reference 8-3 for the discussion of requirements. The following requirements have been developed from the characteristics of the Saturn Ring Rendezvous mission and others presented in Subsection 3.2.

8.1 SAFETY

No change.

8.2 ORBIT OF OPERATION

The orbit of operation, measured in astronomical units (AUs) relative to the Sun, should extend to 50 AU (Interstellar Precursor mission) and in to about 0.3 AU (Mercury Orbiter).

8.3 SOLAR ORIENTATION

For planetary missions, in addition to operation in any solar orientation, the SP-100 should be able to operate for long periods (1-4 years) in essentially the same solar orientation.

8.4 NATURAL RADIATION

Natural radiation for most planetary missions will be less severe than that specified. (See Figures 5-1, 5-2, and 5-3 in Subsection 5.2).

Table 8-1. SP-100 Interactions With NEP Mission Characteristics

SP-100 FEATURES	NEP MISSION CHARACTERISTICS: TARGET	LAUNCH DATE	POWER REQUIREMENT	LIFE REQUIREMENT	PAYLOAD	COST
POWER			X			
CONTROL/DATA						
MECHANICAL						
IR RADIATION						
RF INTERFERENCE						
NUCLEAR RADIATION					X	
CONFIGURATION						
IOC DATE		X				
COST						X
LIFE				X		

Table 8-2. SP-100 Interactions with Spacecraft Subsystems

SP-100 FEATURES	SUBSYSTEM: STRUCTURE	COMMUNICATIONS	POWER	PYRO	COMMAND CONTROL DATA HANDLING	ATITUDE/ ARTICULATION CONTROL	TEMPERATURE CONTROL	MECHANICAL DEVICES	DATA MEMORY	ANTENNAS	SCIENCE	PROPULSION
POWER		X	X		X	X	X		X		X	X
CONTROL/DATA		X	X		X				X			
MECHANICAL	X					X		X				
IR RADIATION							X				X	
RF INTERFERENCE		X			X						X	
NUCLEAR RADIATION	X	X	X	X	X	X	X	X	X	X	X	X
CONFIGURATION	X					X	X	X			X	X
IOC DATE												
COST												
LIFE	X	X	X	X	X	X	X	X	X	X	X	X

Table 8-3. SP-100 Interactions With Spacecraft and Ground Functions

SP-100 FEATURES	COMMUNICATE	GATHER SCIENCE	PROVIDE POWER	TRAJECTORY CONTROL	STORE DATA	PROVIDE A STABLE PLATFORM	MISSION OPERATIONS AND SUPPORT
	FUNCTIONS						
POWER	X	X	X		X	X	
CONTROL/DATA	X				X		X
MECHANICAL						X	
IR RADIATION		X		X			
RF INTERFERENCE	X	X					
NUCLEAR RADIATION		X		X			
CONFIGURATION		X				X	
IOC DATE							
COST							
LIFE							X

8.5 METEOROIDS

The meteoroid environment will be more severe than that specified for those planetary missions that pass through the asteroid belt. (See Subsections 5.1 and 5.4.)

8.6 SPACE DEBRIS

Due to the short time that is spent near the Earth during planetary missions, the debris environment will be less severe than that specified.

8.7 LEO PLASMA

No change.

8.8 DESIGN LIFE

For the missions considered in Subsection 3.2, the full power life is less than 6.3 years. Seven years is a desirable goal.

8.9 DESIGN RELIABILITY

Since the payoff for planetary missions is at the end of the mission, the probability (99%) of full power operation must be specified relative to the end of total system life (see below), not full power design life.

8.10 REPAIRABILITY

No change.

8.11 OPERATIONS

No change.

8.12 DESIGN APPROACH

No change.

8.13 MASS

No change.

8.14 SIZE

No change.

8.15 STS INTERFACE

No change.

8.16 USER ELECTRICAL INTERFACE

No change.

8.17 MAIN BUS

End of life power of 100 kWe must be relative to the total system life (see below), not full power life.

8.18 SECONDARY BUS

Approximately 25 to 50 W_e will be required to extend the long booms of an NEP spacecraft. This power should be in addition to the 300 W_e specified.

8.19 LOAD FOLLOWING

No change.

8.20 USER FAULT TOLERANCE

No change.

8.21 RADIATION DOSE LOCATION

A 4.5-m diameter dose circle is acceptable. The user interface is located 26 m from the reactor.

8.22 FLUENCE AND DOSE

The fluence and dose as specified are barely tolerable with the configuration shown in Subsection 4.4. It is desirable that the dose be lowered to 75 kRad and the fluence to 1 x 10^{12} neutrons/cm^2. The fluence and dose levels should be specified for the total system life (see below).

8.23 THERMAL RADIATION

(Not specifically addressed in this study.)

8.24 ELECTROMAGNETIC INTERFERENCE

(Not specifically addressed in this study.)

8.25 MECHANICAL INTERFACE

(Not specifically addressed in this study.)

8.26 COMMAND/DATA/TELECOMMUNICATIONS

No change.

8.27 NEW REQUIREMENT: INITIAL OPERATIONAL CAPABILITY (IOC) DATE

The earliest IOC date for a NEP planetary mission is 1995. A more conservative date is 2000.

8.28 NEW REQUIREMENT: RECURRING COST

The SP-100 power subsystem recurring cost should be less than $100M (FY 84 dollars) and it is very desirable that it be below $50M (FY 84 dollars).

8.29 NEW REQUIREMENT: REACTOR THROTTLING

For planetary missions with several coast phases, the reactor should be designed to complete A-B-A cycles between operating points A and B as defined below.

A. Power system operating at full rated power.

B. Reactor operating at the lowest level possible, consistent with the ability to resume operation at point A. The radiation flux at point B should be at least a factor of ten lower than at point A.

8.30 NEW REQUIREMENT: REACTOR CYCLE LIFE

The reactor must be designed to undergo at least five A-B-A cycles as described above in order to be consistent with the full power life requirement, the total system life requirement (see below), and the mission requirements of the Saturn Ring Rendezvous and Multiple Asteroid Rendezvous missions. A reactor cycle life of 100 would be very desirable.

8.31 NEW REQUIREMENT: TOTAL SYSTEM LIFE

The total system life must be at least 11.7 years in order to meet the requirements of the Neptune Orbiter and Saturn Ring Rendezvous missions.

8.32 NEW REQUIREMENT: DORMANCY

The power subsystem should be capable of being dormant (defined as operation at point B, above) for at least five years in order to meet the full power life and total mission life requirements of the Neptune Orbiter and Saturn Ring Rendezvous missions.

8.33 NEW REQUIREMENT: CONTINUOUS ACCELERATION

Based upon any NEP mission and allowing for different configurations, the power subsystem should be capable of operation for the full power life while being accelerated in any direction at a magnitude of between 1.0×10^{-4} and 1.0×10^{-3} m/sec^2.

SECTION 9

CONCLUSIONS, FEASIBILITY ISSUES, AND AREAS FOR FUTURE WORK

The following items are the major conclusions of this study.

(1) There is substantial science rationale for and interest in several of the many planetary missions that are enabled or significantly enhanced by nuclear electric propulsion (NEP). (See Sections 2 and 7.)

(2) A feasible overall spacecraft configuration using an SP-100 power subsystem, an electric propulsion subsystem, and a typical planetary instrument payload can be defined using current or projected technical capabilities. (See Figure 4-6.)

(3) Seven years of full power operation, 100 kWe, and 3000 kg are acceptable goals for a SP-100 power subsystem for NEP planetary missions.

(4) The radiation environment is the single most challenging feature of the SP-100 for spacecraft design and integration. (See subsection 5.2.)

(5) The SP-100 power subsystem (as defined by its present baseline requirements (Reference 8-3)) would be compatible with NEP planetary missions if certain options, identified in Reference 8-3, would be included. The adjustments to the baseline SP-100 power subsystem requirements that would make it compatible with planetary missions are (1) a longer life (about 12 years), (2) up to five years of dormancy, (3) the ability to throttle the reactor (in power) down a factor of ten and back to full power at least five times, (4) lower reactor produced radiation, and (5) the capability to survive a more severe meteoroid environment.

The following pages of this section present more detailed conclusion and feasibility issues and areas for future work for each of the major sections of this study. Feasibility issues are those items of mission or system design that have not or cannot be resolved at this time and, by their nature, call into question the viability of the entire mission or system design. Areas for future work are not believed to be feasibility issues, yet are items that need to be investigated; but were not in this study due to resource constraints.

9.1 PLANETARY EXPLORATION SCIENCE OBJECTIVES

9.1.1 Conclusions

- Planetary science objectives are well defined.

- These objectives call for missions that are enabled by NEP or can only be done at a reduced scope or increased risk without NEP.

9.2 NUCLEAR ELECTRIC PROPULSION RATIONALE

9.2.1 Conclusion

- The rationale for using NEP on planetary missions in the context of its enabling characteristics of reduced flight time and increased payload are well developed, understood, and accepted by the planetary science and mission planning communities.

9.3 POTENTIAL NUCLEAR ELECTRIC PROPULSION MISSIONS

9.3.1 Conclusion

- The performance of NEP has been estimated for a sufficient number of potential planetary missions in order to cover the range of science objectives and demonstrate the multimission capability of NEP.

9.4 SATURN RING RENDEZVOUS MISSION

9.4.1 Conclusion

- The mission/system concept presented in this report is not mature. Significant feasibility issues have been identified, and are given below.

9.4.2 Feasibility Issues

- The spacecraft is exposed to a severe radiation environment during this mission due primarily to the SP-100 and the Earth spiral escape. Spacecraft and payload electronic protection is a design challenge.

- The spacecraft is exposed to a severe dust environment during its passage through the E-ring.

- The baseline SP-100 power subsystem, as defined in Reference 8-3, will meet the SRR mission/system requirements if the following SP-100 enhancement options (some of which are defined in Reference 8-3) are added to the baseline SP-100:

  (1) Enhanced meteoroid shielding.

  (2) Reactor cycling capability.

(3) Dormancy capability of ≥ 5 years.

(4) Total power subsystem life ≥ 11.7 years.

9.4.3 Future Work

- A better definition of the radar is needed. This definition should include such items as (1) frequency of operation, (2) pulse power, (3) pulse length, (4) footprint size, and (5) duty cycle.

- The sequencing of the science instruments periods of operation should be defined.

- It should be determined whether the spacecraft should observe the rings of Saturn from above or below.

- Flight time performance enhancement, radiation hazard reduction, and safety/risk improvement are three potential benefits that might occur by elimination of the Earth escape spiral phase. Launch to Earth escape energy by low-cost, reuseable chemical orbital transfer vehicles (OTVs) should be studied.

- The location of the probe's entry into Titan's atmosphere should be defined.

- The time, direction, and release scenario for the Titan probe should be defined.

- The total mission time and full power time need to be adjusted for each mission scenario in order to accommodate the spacecraft mass shown in Table 4-5.

- The performance penalty associated with diverting 6 kWe of power to operate the radar instead of supplying the 6 kWe to the propulsion subsystem during periods of mutual operation should be determined.

9.5 SP-100 POWER SUBSYSTEM

9.5.1 Conclusions

- Seven years of full power operation, 100 kWe, and 3000 kg are acceptable goals for a SP-100 power subsystem for NEP planetary missions.

- The radiation environment is the most difficult SP-100 feature to integrate into a spacecraft design and configuration.

- A reduction of the mass and radiation environment and an extension of the total system life to approximately 12 years or so are the three most desirable modifications to the baseline SP-100 power subsystem from the point of view of satisfying NEP planetary missions requirements.

9.5.2 Feasibility Issues

- The meteoroid environment for the Saturn Ring Rendezvous Plus Radar (SRRPR) mission is more severe than the baseline SP-100 power subsystem requirement, but no different than any other interplanetary mission requirement.

- The baseline SP-100 power subsystem requirements do not include a requirement for reactor cycling, which is necessary to meet the SRRPR mission requirements, and still be consistent with the power subsystem full power life requirement.

- The baseline SP-100 power subsystem requirements do not include a period of dormancy. In order to meet the SRRPR mission requirements and still be consistent with the SP-100 full power life requirement, the SP-100 power subsystem will need to be designed to have a 5-year dormancy period.

- The baseline SP-100 power subsystem total system life is 10 years. The SRRPR mission and other planetary missions require a total system life of 11.7 years or more.

9.5.3 Future Work

- The SP-100 power subsystem operations need to be defined.

- The SP-100 power subsystem user electrical interface needs to be better defined.

- The forces and torques produced by the SP-100 at the user interface need to be defined.

- The command and telemetry needs of the SP-100 power subsystem need to be defined.

9.6 ELECTRIC PROPULSION

9.6.1 Conclusion

- The SP-100 power subsystem user interface and internal power conditioning specifications and their proposed implementation approaches, as presently defined, appear to be generally compatible with the power input requirements of an ion engine power processor.

9.6.2 Feasibility Issues

- The thruster lifetime and power used in this study are within current projections, but still far beyond demonstrated capability.
- The power processor mass and power used in this study are consistent with current projections, but are beyond demonstrated capability.
- It is an open question whether a propellant tank and acquisition devices can be designed to maintain a uniform (or nearly uniform) propellant depth over the 2.2-m diameter as assumed in this study.

9.6.3 Future Work

- The propulsion system specific impulse should be reoptimized in the context of the most recent system and mission design.
- The lifetime of the power processor needs to be investigated to be certain that it is consistent with the mission lifetime requirement and redundancy allocation.
- The feasibility of switching power processors to different thrusters needs to be established.
- The plasma and electromagnetic interference (EMI) environments produced by the electric propulsion subsystem should be determined.

9.7 SCIENCE INSTRUMENT RADIATION SENSITIVITY

9.7.1 Conclusion

- Insofar as the Galileo instruments are representative of planetary science instruments, the study team can say that planetary science instruments, with a few exceptions, are compatible with the SP-100 powered NEP spacecraft configuration developed in this study when properly shielded by modest amounts of neutron and gamma attenuator.

9.7.2 Feasibility Issues

- The energetic particle detector gamma flux tolerance is three orders of magnitude lower than the nominal SP-100 environment and probably will not produce useful results in a low signal regime.

9.7.3 Future Work

- All the science instruments surveyed in this study need to be tested in order to verify their radiation tolerance.
- The effects upon the instrument and its data at its radiation tolerance limit need to be defined.
- The amount of shielding necessary to successfully integrate the instruments with the spacecraft configuration developed in this study needs to be estimated.

9.8 CONFIGURATION AND MASS PROPERTIES

9.8.1 Conclusion

- A feasible overall spacecraft configuration using an SP-100 power subsystem, an electric propulsion subsystem, and a typical planetary instrument payload can be defined using current or projected technical capabilities.

9.8.2 Feasibility Issue

- No dynamic analysis was performed on the spacecraft configuration developed in this study.

9.8.3 Future Work

- The structural integration (cradles, center of mass location, etc.) of the stowed configuration of this spacecraft with the shuttle should be developed.

9.9 ATTITUDE AND ARTICULATION CONTROL

9.9.1 Conclusion

- A feasible conceptual design for the attitude and articulation control system (AACS) for a NEP spacecraft has been developed.

9.9.2 Future Work

- The disturbance environment (including reactor thermal and radiation pressure forces) that the AACS will be operating in should be defined in order that the conceptual design can be verified and the amount of attitude control propellant calculated.

9.10 TELECOMMUNICATIONS

9.10.1 Future Work

- The replacement of the 20-W X-band solid state power amplifiers (XSSPAs) with four- or six-watt XSSPAs like those in the array feed power amplifier (AFPA) should be investigated. This would eliminate the 20-W XSSPAs and the hybrid. The 5-W XSSPAs would be turned on and off by command.

- The addition of beam steering capability to the HGA/AFPA for good HGA pointing should be studied. Only about 0.5° of steering would be required.

- The HGA may be able to be used for the radar as is done on Venus Radar Mapper (VRM). An S-band feed for the radar would be added. In this option the medium gain antenna (MGA) would be mounted on the edge of the HGA.

9.11 CHEMICAL PROPULSION

9.11.1 Future Work

- In order to minimize structural loads and deformations inherent with a centralized RCS approach, a distributed RCS approach should be investigated.

9.12 METEOROID ENVIRONMENT

9.12.1 Feasibility Issue

- The meteoroid environment determined for the SRR mission is more severe than that which the SP-100 power subsystem is now required to meet.

9.13 REACTOR AND NATURAL RADIATION ENVIRONMENT

9.13.1 Future Work

- The radiation environment encountered by the spacecraft during the Earth escape at an inclination of 30° (28.5°) needs to be determined for the SRR mission.

9.14 SATURN'S RINGS

9.14.1 Feasibility Issue

- Whether the spacecraft could survive the dust environment that it would experience as it passed through the E-ring is an open question.

9.14.2 Future Work

- The particle fluence encountered by the spacecraft during its passage through the E-ring should be determined.

9.15 SATURN RING RENDEZVOUS MISSION COSTS

9.15.1 Conclusions

- Based upon the estimate of the development costs for the SRRPR mission, it is concluded that the recurring cost of the SP-100 power subsystem should certainly be below $100M (FY 84 dollars) and preferably $50M (FY 84 dollars), or less.

- Under the current resource allocation and the cost of competing missions, NASA cannot currently afford to start those missions that require or could greatly benefit from NEP such as the SRRPR mission. If, however, the climate of space exploration funding were to change (as predicted by some industrial economic advisers), then several large mission programs, which could afford and greatly benefit from an SP-100 powered NEP, could be on the planning horizons.

9.16 NUCLEAR ELECTRIC PROPULSION MISSIONS PLANS

9.16.1 Conclusions

- If NASA is successful in receiving resources for an augmentation of the planetary exploration core program, it is clear that a NEP mission will be contemplated.

- In any augmented mission plan there are several missions to which NEP can be applied or which may be enabled by NEP.

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APPENDIX

GLOSSARY OF ABBREVIATIONS AND ACRONYMS IN REPORT

A.L.	anomalously large
ACS	attitude control system
A/C	attitude and control
AACS	attitude and articulation control system
ACSR	Asteroid Comet Sample Return
AFPA	array feed power amplifier
ARC	Ames Research Center
ASE	airborne support equipment
ATAC-16	AACS controller (16-bit)
AU	astronomical unit (A.U.)
Ah	ampere-hour
B	billion
BER	bit error rate
BW	bandwidth
CCD	change-coupled device
CM	center of mass
CMD	command
COMPLEX	Committee on Planetary and Lunar Exploration
CDU	command decoder unit
DC	direct current
DDS	dust detector
ΔVEGA	Earth gravity assist trajectory
DSN	Deep Space Network
EPD	energetic particle detector

EMI	electromagnetic interference
EOp	Europa Orbiter/Penetrator
ES	Earth storable
g	gram
G&A	general and administrative
GISS	Goddard Institute for Space Study
GLL	Galileo Project
GSOP	Galilean Satellite Orbiter/Probe
HEF	high efficiency
HGA	high gain antenna
HTC	high thrust chemical
I/O	input and output
IOC	initial operational capability
IP	IPPACS platform
IPPACS	integrated platform pointing and atitude control subsystem
IR	infrared
IRU	inertial reference unit
kWe	kilowatt electrical
LASP	Laboratory for Atmosphere & Space Physics
LEO	low Earth orbit
LGA	low gain antenna
LeRC	Lewis Research Center
M	million
MAG	magnetometer
MGA	medium gain antenna
MMH	monomethylhydrazine
MSR	Mars Sample Return

NASA	National Aeronautics and Space Administration
NEP	nuclear electric propulsion
NIMS	near infrared mapping spectrometer
NOP	Nepture Orbiter/Probe
NSO	nuclear safe orbit
NTO	nitrogen tetroxide
OMV	orbital maneuvering vehicle
OTV	orbital transfer vehicle
PLS	plasma science
PM	payload module
PO/P	Pluto Orbiter/Penetrator
PPR	photopolarimeter radiometer
PPU	power processing unit
PWS	plasma wave science
RCS	reaction control system
RDM	radiation design margin
RF	radio frequency
RS	radio science
RTG	radioisotope thermoelectric generator
SAIC	Science Applications International Corporation
S/C	spacecraft
SCS	satellite control system
SEEGA	solar electric Earth gravity assist
SEP	solar electric propulsion
SP	space power
SRR	Saturn ring rendezvous
SRRPR	Saturn ring rendezvous plus radar

SS	space storable
SSEC	Soar Sytem Exploration Committee
SSI	solid state imaging science
SSPA	single sideband power amplifier
STS	Space Transportation System
TBD	to be determined
TBST	target body star tracer
3-D	three dimension
TM	thrust module
TMU	telemetry modulation unit
TV	television
TWTA	traveling wave tube amplifier
UV	ultraviolet
UVS	UV spectrometer
VGR	Voyager Project
VRM	Venus Radar Mapper
XPNDR	transponder
XSSPA	X-band solid state power amplifier

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